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Report No [REDACTED]
Page ii



CORONA 83 [REDACTED]
SYSTEM PERFORMANCE/DESIGN
REQUIREMENTS (TSSM)

Approved: [REDACTED]

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[REDACTED] SYSTEM PERFORMANCE/DESIGN

REQUIREMENTS

MASTER SYSTEM SPECIFICATION

JUNE 1966

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1.0 SCOPE

This specification covers the requirements for the Corona J3/201 Satellite Search Surveillance System which shall be capable of obtaining high resolution photography of specific geographic areas. The Satellite Vehicle flight system and ground support required for launching of the vehicle, its orbital control, and data retrieval shall also be specified. This document is intended to specify the configuration of the Program as of July 1967.

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

MIL-E-6051C	Interference Limits and Methods of Measurements, Electrical and Electronic Installation in Airborne Weapons Systems and Associated Equipment.
MIL-I-26600	Interference Limits and Methods of Measurements, Equipment.
MIL-Q-98581	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 Preparation
MIL-STD-826	Electromagnetic Interference Test Requirements and Test Methods
MIL-STD-150A	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

CP 00066

Production and Acceptance Specification
SLV-2G Thrust Augmented Thrust Booster

Lockheed Missiles and Space Company

LMSC 1324217

F/L Mech. Interface

LMSC 1419071

Program [redacted] Satellite System Specification

LMSC 1414878

Detail Specification, SS01B and SS01B Standard
Against Vehicle

LMSC 1418333

Recovery Specification

LMSC 1417676

Detail Specification, Model 39205 Vehicle,
Program [redacted]

EO J3-001

Requirements Specification, J-3 System Payload

LMSC T3-6-002

General Specification for Payload Qualification
and Acceptance

LMSC T3-5-023

[redacted]

LMSC-817971

Special Programs Against Secondary Payload

LMSC 1418956

Interface Control Specification between SLV-2G/
SS01B Vehicle

LMSC 1376862

SLV-2G/SS01B Electrical Interface

LMSC 1376863

SLV-2G/SS01B Mechanical Interface

Program Requirement Document -

Program [redacted] National Range Division P 9026,
Western Test Range Serial No. 00504, Headquarters,
WTR, Point Mesa, California. (prepared as
LMSC B079854, 1 May 64)

Program [redacted] Orbital Requirements Document -

6595 ATW, WTR 4-17 (Prepared as
LMSC B079854, 1 May 64)

Bell Telephone Laboratories/Western Electric Company

BTL 6738648

Guidance Equations Specification, Ground
Guidance Computer Programs

BTL 6734173

Electrical Acceptance Test Specification,
Series 600, Missile-Borne Guidance Equipment.

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Program [redacted] Recovery Requirements Document

Program Support Plan, Program [redacted] 79012 Western Test Range

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

Program [redacted] Recovery Support Plan

Thorad/Agena Range Safety Report for the Western Test Range, IMSC A 815752

Test Operations Order No. 64-3, Program [redacted] 694 ATM, May 1964

Recovery Group Operations Plan No. 1-64, Program [redacted] Hq. 694 Recovery Control Group, 694 ATM AFSC

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

Flight Termination System for the Agena Space Booster Launched on the Western Test Range, IMSC A 798744

Safety Procedures Manual, IMSC Operations at Vandenberg Air Force Base - IMSC 220580 Rev. 1, 14 March 1964

Launch Stand Safety Plan, IMSC 224455 B

AFR 205-23 Special Security Procedures Projects

Count Down Manual IMSC 226785, WAFB Complex 75

Count Down Manual IMSC 226791, PALC-1

Program [redacted] Explicit Guidance Equations

Air Force Western Test Range Manual 127-1 dated 25 Feb. 1966

Bulletins

ANA Bulletin 438A

AGE Control for Synthetic Rubber Parts

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3.0 REQUIREMENTS

3.1 Performance

The objective of the Corona J-3 [REDACTED] System is to obtain terrain photographic data of specified geographic areas from a satellite vehicle for search and surveillance purposes. In accomplishing this 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 a pre-selected ocean location.
- E. Provide necessary telemetry, tracking, and commands - both vehicle and ground equipment.
- F. Control of equipment temperatures throughout mission phases.
- G. Stable re-entry with environment protection of recovery data recovery vehicles.
- H. Recovery of data capsules during descent or after water impact.
- I. Assembly, checkout and launch functions by Aerospace Ground Equipment.

The Corona J-3 [REDACTED] System shall be capable of obtaining stereoscopic and monoscopic photos by utilizing panoramic cameras operating in orbit at an altitude range of 80 to 200 nautical miles. The panoramic camera photographic scan angle shall be 70 degrees, yielding a swath width of 130 nautical miles at a Satellite Vehicle altitude of 90 nautical miles. For a single mission at 90 nautical miles altitude the normal panoramic film capacity represents a total stereo ground coverage of 7.7 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.

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The panoramic camera subsystem shall be capable of being programmed for the desired portion of the ground track on any given orbit. A Dual Improved Stellar Index Camera shall also be utilized to obtain terrain and stellar photography. This camera shall be capable of running in a slave mode to the Panoramic Cameras or on an independent basis. The system shall be capable of performing missions of 14 days duration with early call down capability. Dual recovery vehicles each having a capacity for one half the total film load of both the panoramic and DIIC cameras 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. The system will utilize a Thor Augmented Thorad Boost Vehicle (SEV-2G) and a modified standard Agena (80-1B)* for launch from the Western Test Range (WTR) into 60° to 110° orbit. In addition, to performing the function of second stage boost vehicle, the Agena also serves as the Satellite Vehicle which supports the photographic payload and re-entry vehicles. The re-entry vehicles containing the photographic film shall be recovered by air or sea retrieval over water, or alternatively, to be recovered by air or sea retrieval. This is 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-surveillance 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. Tracking stations of the SCF that are

* 801B and 8801B used interchangeably.

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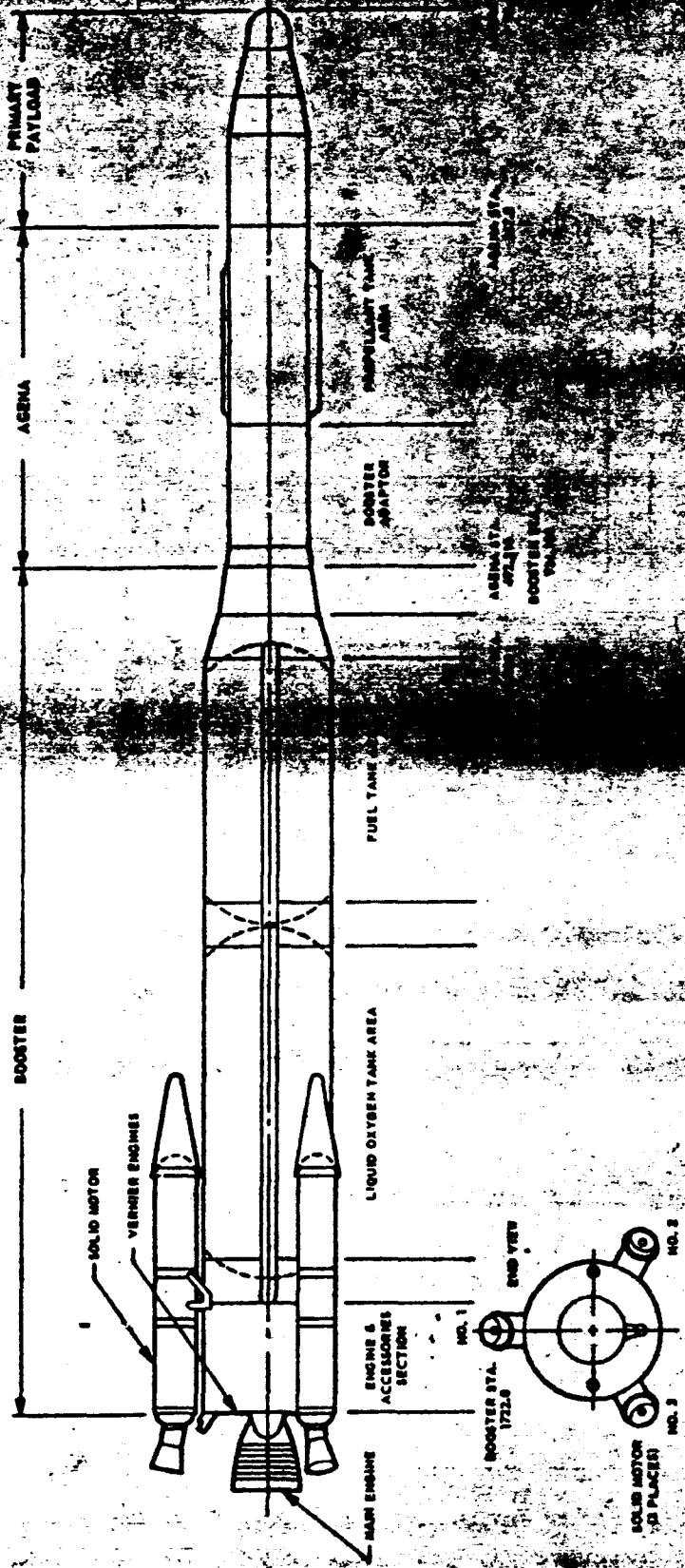


FIGURE 1 SATURN VEHICLE CONFIGURATION

utilized to perform this function are the Tracking Station, the Tracking Station, the Tracking Station, the Auxiliary Station, the Tracking Station, the Tracking Station, and the Tracking Station. Other stations which may or may not be a part of the SCF may support operations as necessary on an individual flight or flight series basis if required. The maximum number of consecutive 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 shall normally be programmed on an orbital programmer. The programmer shall be capable of real time stored programs for 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, eccentricity, 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 3.5 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. Primary data for ephemeris prediction shall be obtained by the use of a PRELORT Radar and a satellite borne transponder.

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 IIR, 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 STU 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 by the Flight Test Field Director (FTFD), Air Force Satellite Control Facility (AFSCF) or direction by the Program Directorate for the vehicle and the Payload Sub-Assembly Project Office (PSAPO) for the payload. In the case of abnormal flight conditions or anomalies of the vehicle and/or payload, commands shall be authorized by the Program Directorate of the Program Directorate.

Vehicle commands transmitted by the SCF will be based upon ephemeris data obtained and reduced by equipment presently in use in the SCF. Payload command will be based on the payload data available generated by the PSAPO. Available proven software and procedures shall be utilized to the greatest extent possible without compromising system goals. Mission preparation time shall be compatible with the frequency of Program Launches.

B. Launch Operations

Program 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 complex will be utilized to assemble, check out and launch the SLV-2G/SOLB boost vehicle.

- | | |
|---------------------|---------------------|
| Launch Complex 75-3 | Pads Number 4 and 5 |
| PAIC-1 | Pad Number 1 |

Program 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 range of a land based station. In addition, the facilities of Vandenberg

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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 SLV-20/SOLB 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 requires that at least one successful mission be completed every 30 days. Maximum launch rate shall be two per calendar month, with an expected normal launch rate of one per month. A turn-around time of 15 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, with accepted degradation in performance and reliability 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 70 percent probability of success. The standby vehicle will remain fixed throughout the period this requirement is in effect, maintenance for this particular vehicle 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.

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The recovery sequence is divided into two phases: re-entry and recovery. The re-entry phase shall start with the initiation of a programmed command. Following this command, the Satellite Re-entry Vehicle (SRV) beacon and telemetry shall be turned on to permit detection, tracking and data recording of the re-entry sequence. The Satellite Vehicle shall be pitched over a nominal 120 degrees from the local horizontal, the film cut, and the SRV separated from the Satellite Vehicle. Descent shall be achieved by means of spin jets, a retro-rocket, and de-spin jets. Immediately after de-spin, the thrust cone (mounting platform for the rocket and de-spin jets) shall be separated from the re-entry vehicle. The recovery phase shall consist of the deployment of the parachute system, ejection of the ablative shield, and activation of a flashing light at approximately 50,000 to 60,000 feet.

The recovery force will consist of aerial recovery aircraft equipped with electronic detection and direction finding equipment. Minimum of three recovery aircraft shall normally be deployed in a North Pacific area, with the impact predictor provided by the WTR ships. Special air retrieval gear to insure safe and secure recovery during 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 shall be rendered by Air Rescue Aircraft with para-rescue capability, weather reconnaissance aircraft, and land-based helicopters for sea surface recovery.

Two SRV's shall be carried by each Satellite Vehicle. Recovery of the first SRV (capsule) will normally be accomplished in from one to ten (10) days after launch and the second SRV (capsule) will be recovered in from one to fourteen (14) 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 seven days duration, however a capability shall exist to cut the film in the A SRV and continue the mission in the B mode with subsequent A recovery at a later time.

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

Subsequent to recovery, capsule handling and disposition will be in accordance with the directives of the Special Projects Directorate. Until such time as the designated courier is able to assume physical custody, the Commander AFSCF-RCG will be responsible for the capsule's physical and security safeguarding per designation by the Commander AFSCF. Upon assuming physical custody, the courier shall be responsible for capsule handling, and security. Transportation and delivery to be designated processing center is the responsibility of the SPO.

3.1.1.1.1 Mission Particulars

A. Orbital Elements - For a particular flight the orbit parameters will be specified on the basis of payload weight, capabilities, and range available from the launch vehicle employed. The parameters are the orbital period, perigee altitude, location of perigee and orbital inclination usually in the aforementioned order. 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 SLV2G/SO1B 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 110 degrees)
- 2. Range of perigee altitude: 80 to 200 n.m.
 - * (Most probable perigee altitude: 85 to 110 n.m.)
- 3. Range of orbital period: 88 to 91.5 minutes
- 4. Range of perigee location: 90°N to 90°S latitude
 - * (Most probable perigee location: 20°N to 60°N latitude)

Vehicle structural limitations may preclude flying all possible combinations of the above parameters. Hence, at inclinations below 88 degrees, the injection altitude shall be the lowest value compatible with SLV2G/SO1B structural capabilities. Mission duration shall be dependent upon the orbit inclination and perigee altitude selected.

* Note: 8 day lead time for mission preparation applies to these ranges only.

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

- 1. Inclinations: Plus
Minus
 - 2. Perigee Altitude: Plus
Minus
 - 3. Period: Plus
Minus
 - 4. Argument of Perigee:
 - (a) For eccentricities of 0.008 or less: Plus
Minus
 - (b) For eccentricities greater than 0.008: Plus
Minus
- } TO BE DETERMINED

B. Ascent Requirements

Program [REDACTED] launches using the SLV-3/SCD vehicles from the following launch sites at Vandenberg AFB:

Complex	Pad No.	Pad Azimuth from North	Geodetic Latitude	Longitude	Elevation above SL
75-3	4	181.48°	34.757302°	120.62917°	125 ft.
	5	218.50°	34.756153°	120.62522°	160 ft.
PALC-1	1	223.53°	34.643653°	120.59903°	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 88 degrees, a yaw (dog-leg) maneuver will be required because of range safety limitations on launch azimuth. Hence, for trajectories which require orbit inclination angles of less than 88 degrees the dog-leg maneuver must be accomplished after the predicted downrange impact has passed the critical range safety boundary.

The Launch Window shall normally 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 vehicle thermal considerations. 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 and horizon sensor ascent look angle.

Ascent Sequence of Events for a typical Program [REDACTED] mission is as follows: This sequence is representative of a 90 degree inclination orbit with injection at 100 nautical miles altitude and a period of 92 minutes.

<u>Event</u>	<u>Time (Sec)</u>	<u>Down range distance (N.M.)</u>
Launch	0	0
Solid Motor Burnout	50	51
Solid Motor Separation	162	9.8
Booster Main Engine Cutoff	215	123.7
Vernier Engine Cutoff	227	142.5
Booster Separation	231	168.0
Optical Door Ejection	233.5	
Stage II Engine Ignition	234.5	
Solid Motor Ignition	235.5	
Booster Impact	242.5	
Stage II Engine Cutoff (orbit injection)	483	869.0

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 Special Projects Directorate. For the SLV20/SOLB 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 and/or Lifebeat timer.
3. Satellite vehicle velocity meter and radio guidance antenna.
4. Satellite vehicle orbital programmer.
5. Stage I Booster autopilot programmer.
6. Ground Command Guidance Computer.
7. Battery and Control gas loading.
8. Range Safety flight data.
9. Solid motor drop time.
10. Payload Delay Settings

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 8 days. This assumes [REDACTED] only as the standard aft payload configuration. Any special research payload may require additional programming preparation.

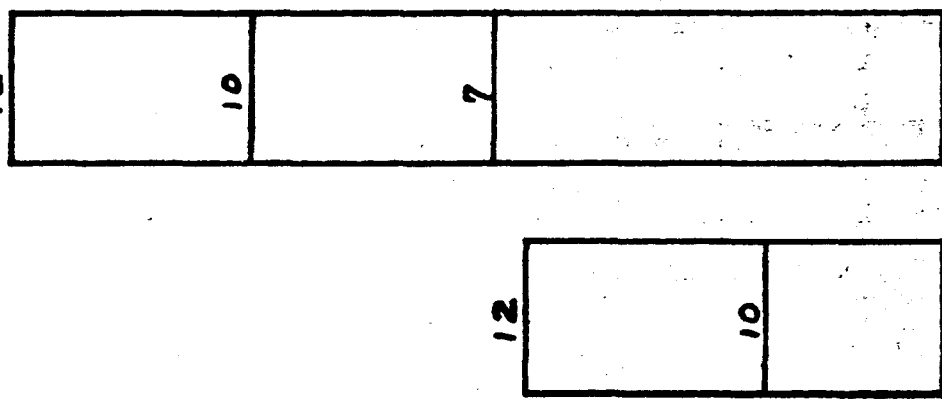
C. Mission Performance

Nominal Mission Duration shall be 14 days of active operation. For any particular mission, the number of 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 trajectory, and the vehicle performance parameters. Figure 2 illustrates the nominal Orbital Weight Capability of the system using SLV2G boost vehicles launched from WAFB 75-3, Pads 4 or 5. Perigee altitude shall be nominally 83 - 100 nautical miles, and the mission parameters are given in terms of orbit plane inclination and period, thus representing the number of days of orbit and the ground track initially covered. Aspect of the ground track is termed orbit synchronization.

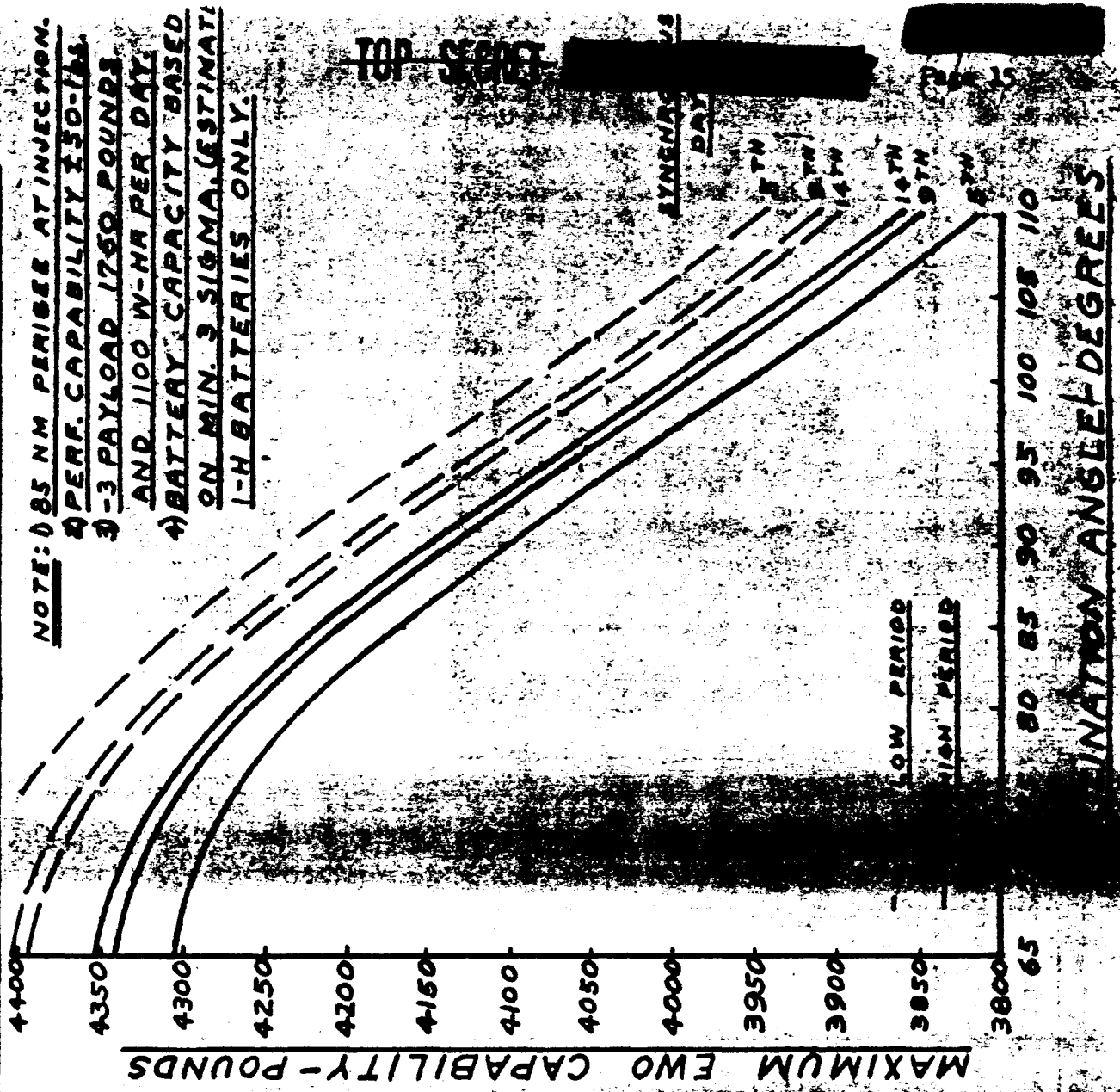
Orbit Sustainance and Orbit Maneuvering Control are not required provided that mission requirements for orbit plane inclination are achieved and that ground track synchronization can be maintained for the specified mission duration. For inclinations between 70 degrees and 91 degrees, mission requirements 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 (low eccentricity) 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

VEHICLE ACTIVE LIFE CAPABILITY-DAYS
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FIGURE 2
ESTIMATED PERFORMANCE CAPABILITY - VAFB (U)



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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 99.0 percent probability that the satellite vehicle altitude shall not decrease below 80 nautical miles during the active mission phases. Efforts shall be directed toward determining means of extending the system capability to attain near-circular (low eccentricity) orbits of approximately 85 nautical miles altitude for the highest orbit plane inclinations consistent with SLV-20/SOLB vehicle performance. Implementation of changes required to attain extended capability shall be approved by the Program Directorate.

A system shall be implemented that shall utilize small solid rocket motors that may be fired in the boost or deboost direction. To achieve a deboost capability the vehicle may be pitched to align the thrust axis in the required direction. These rocket motors shall have a nominal impulse of 2250 pound sec. per motor.

Deboost and Re-entry shall be achievable on all passes. Recovery shall be possible on South to North passes from the Hawaii recovery zone. The vehicle primary guidance and control subsystem shall be capable of recovery 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 back-up attitude control capability shall be provided in the satellite vehicle which will allow at least one recovery on the North to South passes if the primary subsystem should fail.

The primary latitude for the Hawaii recovery zone shall be 24 degrees North on North to South and 18 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.

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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 mission.

Orbit Inclination	85 degrees
Eccentricity	0.022
Period	91.86 minutes

Guidance & Control Subsystem	Direction	Dispersions (n.mi.)		
		Up-range	Down-range	Cross-range
Primary	N to S	73	102	+ 10
Primary	S to N	141	200	+ 16
Back-up	N to S	104	176	+ 13

Abort of the launch or the orbit

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. Similarly, 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 an orbital phase abort, the payload shall be recovered (if possible). If recovery cannot be effected, a self-contained SRV timer shall be started with the receipt of the first recovery command (ARM), shall not perform the normal reverse thrust but shall perform recovery functions such as thrust cone eject, ablative shell off and parachute deployment at a later time than would be performed in a normal recovery. Since these events shall occur in orbit, the subsequent SRV re-entry due to drag shall cause the SRV (with payload) to break up.

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 System.

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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 0.70
- 2. Launch, Ascent, and Orbit Injection 0.95
- 3. Orbital mission (8 days active) 0.84
 - (9 days active) 0.82
 - (10 days active) 0.80
 - (11 days active) 0.79
 - (12 days active) 0.77
 - (13 days active) 0.75
 - (14 days active) 0.74
- 4. Re-entry and Recovery (each ERV) 0.85

On-Orbit Disposal of the satellite vehicle and payload shall occur by the natural process of orbital decay 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 Responsibility

A. Mission Responsibility - The management and mission responsibility relationships for [REDACTED] System are as follows:

Program Management - The Director, [REDACTED] as SPD, is responsible for an overall system engineering (including master system specifications) and system integration (including major sub-system interface specifications); overall system master planning, programming, and budgeting; assembly and checkout of the system at the launch pad; launch and mission operations; capsule recovery and delivery of film to DFRD-designated processing facilities.

In addition, the SPD is responsible for: the thrust-assisted THOR and THORAD boosters; the Agena booster/spacecraft; procurement of the DISIC; the acquisition and operation of system assembly (excluding the LMBC-AP facility) and launch facilities; on-orbit command and control facilities; and capsule recovery forces and equipments.

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The Director, PSAPO, is responsible for total payload sub-assembly development, production (excludes procurement of the DIBIC) and test; the provision of the software support to the Satellite Operations Center before, during, and after missions; operation of the IMSC-AP facility; and adherence to master system specifications; interface specifications; and master project plans established by the SPD.

By definition, the CORONA Payload Sub-assembly includes the CORONA cameras, the HI and/or DIBIC, film transport mechanisms, the RV's, supporting structure and shell, and those other items normally installed and tested at the IMSC-AP facility.

In addition, the PSAPO is responsible to the SPD to assist and manage as appropriate, those Payload Sub-Assembly system assemblies and pre-launch activities at Vandenberg AFB and to certify the appropriate time that the Payload Sub-Assembly is ready and to act as the Payload Sub-Assembly Liaison to the SPD.

General System Engineering [REDACTED] as provided for in applicable contracts.

Launch Vehicle Contractor

The Launch Vehicle Contractor shall provide the Stage I Booster vehicles, and will also provide all services necessary to checkout and launch the vehicle. Douglas Aircraft Company is the Launch Vehicle Contractor for the SLV-2G Thrust Augmented Thorad used for Program [REDACTED]

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Ascent Guidance Contractors

The Guidance Contractors 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 Univac Division of Sperry Rand 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 and through SRV separation. 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 Agona SSQIB used for Program [REDACTED]

Payload Contractor

The Payload Contractor shall provide the payload, [REDACTED] and shall integrate the installation of camera equipment, Satellite Vehicles, and associated components necessary to operate the payload. Camera equipment and SRV's will be supplied to the Payload Contractor as Government Furnished Equipment. Lockheed Missiles and Space Company provides the J-3 payload integration and engineering for Program [REDACTED]

Panoramic Camera Equipment Contractors

Panoramic cameras, film cassettes, and associated photographic equipment, except film, shall be provided for the [REDACTED] System by the Panoramic Camera Equipment Contractor. This equipment will be supplied GFE to the Payload Contractor, however, the Camera Equipment Contractor will furnish field service support under the direction of the PEAPO. Itek Corporation provides the Panoramic J-3 camera equipment for Program [REDACTED]

Dual Improved Stellar Index Camera (DISIC) Contractor

The DISIC, film cassettes, and associated photographic equipment, except film, shall be provided for the [REDACTED] System by the DISIC Contractor. This equipment shall be supplied GFE to the Payload Contractor. Fairchild Camera and Instrument Corporation (FCIC) is the DISIC Contractor.

Film Contractor:

The film contractor shall be responsible for manufacture of the film,

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necessary checkout, and loading of the supply spools and processing of the recovered film. This equipment shall be supplied GFE to the Payload Contractor. Eastman Kodak is the Film Contractor.

Re-entry Vehicle Contractor

The Re-entry Vehicle Contractor shall provide the Satellite Re-entry Vehicles (SRV's) comprising the data capsule, ablative re-entry heat shields, retrieval aids and associated components required for SRV operation. This equipment will be supplied as GFE to the Payload Contractor. General Electric is the Re-entry Vehicle Contractor for Program [REDACTED]

Launch Operations

The 6595th 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 pre-launch and launch operations for this program at the Vandenberg AFB launch sites.

Orbital Operations

Orbital operations shall be conducted from the [REDACTED] Satellite Control Facility operating from the Satellite Control Center, Sunnyvale, California. The AFSCF shall be responsible within the direction of the [REDACTED] Program Directorate and the PSAPO 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 AFSCF Recovery Control Group as designated by the Commander, AFSCF.

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 pre-launch tests to verify a condition of flight readiness for this vehicle. Additionally, the Integrating Contractor shall be responsible for the preparation of pre-flight trajectories, ascent guidance target tape, orbital programmer tapes, range safety data and integrated documentation required to support the launch and vehicle orbital operations for this program.

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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 cases, spares shall pass applicable acceptance tests prior to installation on a Program [REDACTED] vehicle. The contractors also shall define controls for [REDACTED] of spares. Such controls shall [REDACTED] 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

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FIGURE 3 DELETED

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[REDACTED]



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PHASE
ITEM

MISSION PREPARATION

ASCENT GUIDANCE

PROGRAMMED ORBITAL CMDS

CAMERA EQUIPMENT

RE-ENTRY VEHICLES

PAYLOAD SUBSYSTEM

STAGE II VEHICLE

STAGE I BOOSTER

LAUNCH CONFIGURATION COMPLETE

SATELLITE VEHICLE

RECOVERY CAPSULES

FACTORY OPERATIONS & MISSION PLANNING

LAUNCH BASE OPERATIONS

ORBIT

ORBITAL ELEMENTS

ASCENT TRAJECTORY

VEHICLE HDWRS SETTINGS

RANGE SAFETY

SOLID CORP & COMB TIME

ORBITAL ELEMENTS

ORBITAL PROGRAMMING

ORBITAL ELEMENTS

FABRICATE

ASSEMBLY

TEST & ACCEPT

SHIP

FABRICATE

TEST & ACCEPT

SHIP

FABRICATE P/L SECTION

INTEGRATE EQUIP INTO P/L

SHIP

STD. VEH. (GFE)

PREPARE

TEST & ACCEPT

SHIP

STD. VEH. (GFE)

TEST & ACCEPT

SHIP

TEST & ACCEPT

SHIP

AFWTR PLOT BOARDS

GUIDANCE COMPUTER

TAPE PREP & C/D

BOOSTER ON STAND

MATING & PRE-LAUNCH TESTS

SOLID MOTOR BOOSTERS

COUNTDOWN & LAUNCH

ORBIT INJECTION

Major operations flow diagram

FIG 4

PERATIONS

RECOVERY OPERATIONS

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ORBIT OPS, PLNG
AND DOC.

RCVRY, OPS, PLNG.
AND DOC.

A MISSION

TRACKING, TIM
CMMD VEH. & P/L

START
DE-BOOST

P/L
WITCHOVER

TRACK, TIM
CMD

START
DE-BOOST

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RE-ENTRY

TRACK

INTERCEPT
RECOVER

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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 shall consist of an Agena SS01B, orbiting vehicle containing the payload equipment and re-entry vehicles. The satellite vehicle subsystem shall provide all necessary functions to fulfill the space-borne mission requirements.

3.1.2.2.2 Launch Vehicle Subsystem

The Launch Vehicle Subsystem shall consist of a Thrust-Augmented-Thorad SLV-2G, for the first stage booster vehicle with the Agena SS01B, serving as the second stage booster. Ascent guidance shall be provided by on-board autopilots and by Radio-Command Guidance utilizing radar tracking, and a ground based computer to compute the guidance commands during the guidance interval of the ascent phase. 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 Specifications 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.

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SYSTEM PERFORMANCE AND DESIGN REQUIREMENTS MASTER SYSTEMS SPECIFICATION

SYSTEM TEST PLAN PROGRAM

PAYLOAD TEST PLAN J3 SYSTEM
LMSC T3-6-032

VEHICLE TEST PLAN AGENA MODEL 39-205

VEHICLE TEST PLAN THORAD SLV-26
DAC -

FACILITIES MASTER PLAN

PROGRAM REQ. DOC. NATIONAL RANGE DIV.
WTR S/N 00504

PROGRAM SUPPORT PLAN
WTR F 9012

RANGE SAFETY REPORT THORAD/AGENA FOR WTR

COUNTDOWN MANUAL PROGRAM

75-3
PAGE 1

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ORBITAL REQ. DOC PROGRAM

ORBITAL SUPPORT PLAN PROGRAM

TEST OPNS. ORDER PROGRAM
AFSCF ATW NO. 64-3

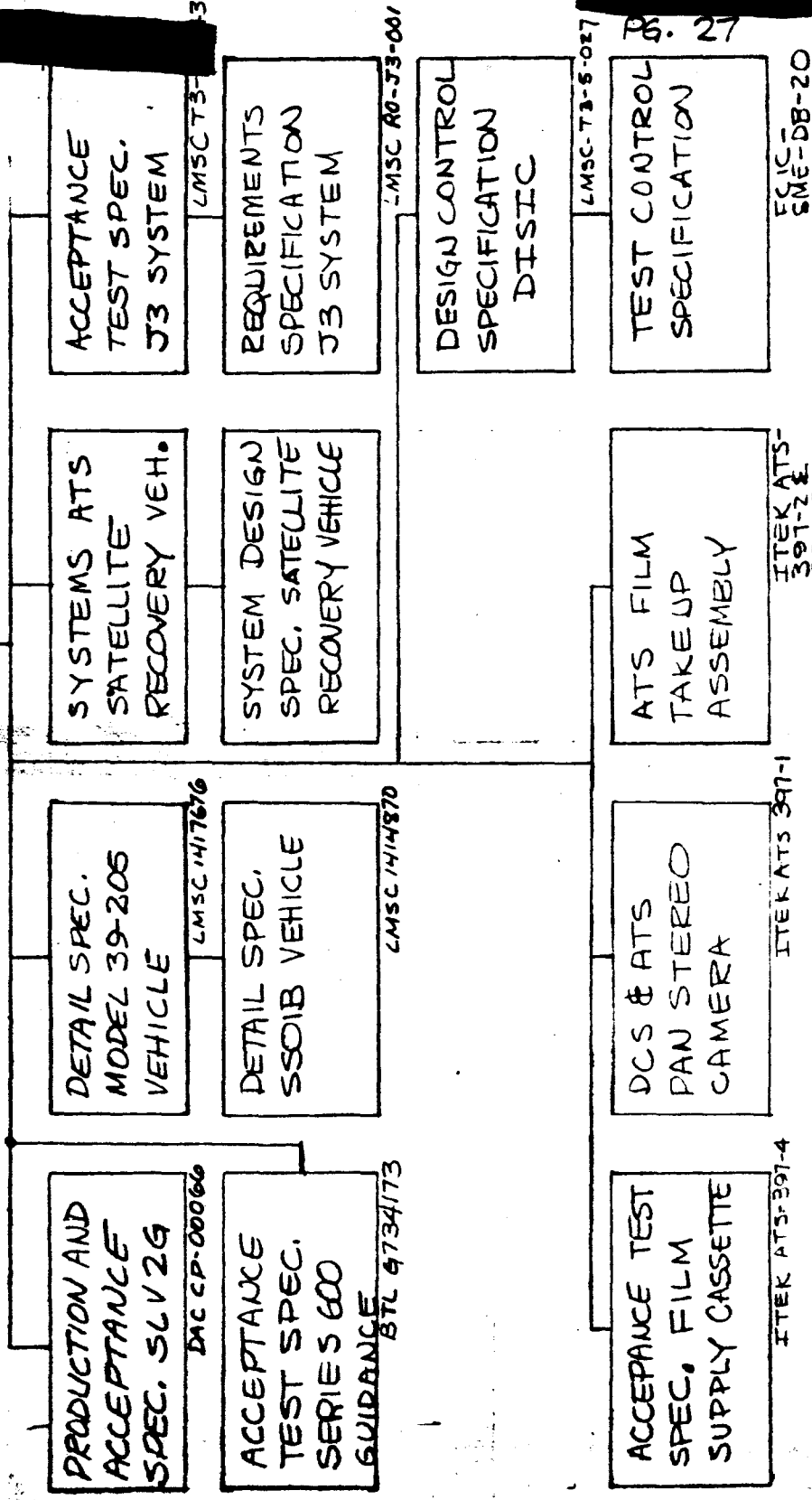
RECOVERY GP. OPS PLAN PROGRAM
AFSCF-RCG NO 1-64

P626

FIGURE 5 OPERATION REQUIREMENTS & PLANS

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CORONA J3 SYSTEM
PERFORMANCE AND DESIGN
REQUIREMENTS
MASTER SYSTEMS SPECIFICATION



8. 27

FIGURE 6 SPECIFICATION TREE - MAJOR END ITEMS

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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 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

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categories: i.e., Reliability Design Objectives and Reliability Flight Goals. The latter category was discussed in Section 3.1.1.1.1 of this document.

Reliability design objectives shall be defined as those reliability numbers assigned to the functional subsystems for the purposes of design apportionment to respective vehicle subsystem and component levels.

The reliability figures given below represent the calculated probability that all 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 B01B (orbital mode, 14 days maximum active operation) Design Objective .82
- B. Payload Equipment (14 day operation) Design Objective .85
- C. Re-entry Vehicles Design Objective .85

3.1.3.1.2.2 Launch Vehicle Subsystem

A. Stage I Booster Vehicle (SLV-2G)

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 0.97 is to be utilized.

B. Stage II Booster Vehicle (Agena B01B), ascent mode)

Design Objective .97

C. Radio Command Ascent Guidance

The reliability design objective of the radio command guidance system is outside the scope of this specification. For the purpose of reliability apportionment, a figure of 0.985 shall 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

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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 mission objectives be reduced by a failure to properly repair or maintain essential equipment.

Payload Reliability

Payload reliability shall be achieved by careful observance of standard design and manufacturing practices. A formal reliability program is not a part of the payload development plan.

3.1.3.3 System Environments

3.1.3.3.1 Prelaunch Environment

The pre-launch 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 covering will be used 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 associated with the pre-launch 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-883 (USAF).
- E. Smoking - Exposure to radiant energy equivalent to exposure to the test outlined in Procedure 505 of MIL-STD-883 (USAF).

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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 dependant 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 Booster Station 151

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Maximum Ground Wind Conditions - Including Gusts*						
Condition	Wind Velocity - Knots					
	(A)		(B)		(C)	
Launcharr	(P)	(U)	(P)	(U)	(P)	(U)
Booster & SS01 B Empty w/o Solids		14		44		75
Booster & SS01 B Empty with Solids		25		44		75
Booster with Solids Empty SS01B Fueled		29		42		75
Booster Fueled (with Solids) & SS01B Fueled	43	37	43	37	75	75

3.1.3.3.2 Launch Ascent Environment

The ascent phase shall include that period of time from the ignition of the Stage I Booster through injection of the Satellite Vehicle into orbit.

A. Agens Environment: The Agens shall undergo an environment as described in IMSC 6117. The testing to be done to determine hardware flight qualification shall be to the B or D version as determined by specific contract negotiations.

B. Payload Environment: All payload elements shall be qualified to withstand a launch and orbital environment as described in IMSC T3-6-002.

C. 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 Contractor's design shall ensure that all equipment intended to operate with a zero differential pressure across various components, shall be properly vented to accommodate the reduction in ambient pressure. Equipment intended to operate under a pressurized environment shall be adequately sealed.

D. Winds Aloft: The winds amplitude, rate of shear, duration of shear and atmospheric density are factors that may cause the ascent vehicle structural, control or range safety capability to be exceeded. Prior to launch, the Integrating Contractor shall compute the wind shears utilizing the winds aloft

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data from standard Rawinsonde observations, and provide the Launch Test Wing with percentages of structural and control capability. Standard observations will be accomplished at 0415 Greenwich time and subsequent 6 hour intervals. If sufficient safety margins cannot be attained because of the winds aloft, the launch shall be delayed.

E. Atmospheric Density Model: The ARD Model Atmosphere 1962 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.

F. Sound Pressure Levels (SPL): The sound pressure level that is expected to exist on a SLV-20/8801B vehicle are tabulated below. These SPL are expected to last no longer than ten seconds, and are measured at the base of the booster adaptor.

<u>Frequency Range</u>	<u>Exterior SPL (db)</u>	<u>Interior SPL (db)</u>
18.8 - 9600	158	145
18.8 - 37.5	140	125
37.5 - 75	143	125
75 - 150	146	128
150 - 300	149	133
300 - 600	152	137
600 - 1200	152	138
1200 - 2400	149	139
2400 - 4800	145	139
4800 - 9600	139	136

Reference level is .0002 dynes per square cm.

G. Axial Acceleration [redacted]: The axial acceleration listed below is expected to occur approximately * seconds before MECO and be of a 0 to * peak acceleration with a frequency range of * to * Hz.

3.1.3.3.3 Orbit Environment

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

* NOTE: To be determined.

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- A. Vacuum: 1×10^{-8} mm of Mercury
- B. Solar Radiation: 445 BTU/ft²hr (nominal)
- C. Earth Shine: 68.7 BTU/ft²hr (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.mi.
- F. Atmospheric Density: The Lockheed-Jacchia or the LDENSITY atmosphere models shall be used. The LDENSITY model is defined in LMBC 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 1962 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 Satellite Control Test Wing for optimizing the re-entry and impact position.

B. Air Retrieval: At the time of parachute deployment, the capsule may be oscillating in a 20 degree cone and be rotating at a rate of 15 rpm. The parachute system shall be capable of proper deployment under these conditions, and shall provide a rate of descent of less than 29.5 ft./sec. under standard atmospheric conditions, at 10,000 ft. altitude above Mean Sea Level (MSL). For weights of 120 to 230 lbs. suspended under the main canopy, the chute/capsule shall be capable of sustaining retrieval loads by aircraft traveling at 135 knots (indicated) at a maximum altitude of 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 without capsizing in sea states of 3 or less for longer than 55 hours and less than 85 hours. (See

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state as defined in 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 tracks or special trailers as required. Payload equipment shall be capable of being transported by aircraft from the Contractor's plant to the Contractor's facility. During all periods of transportation, positive security safeguards shall be provided to physically protect System flight equipment and preserve the required level of secrecy concerning its use in the mission.

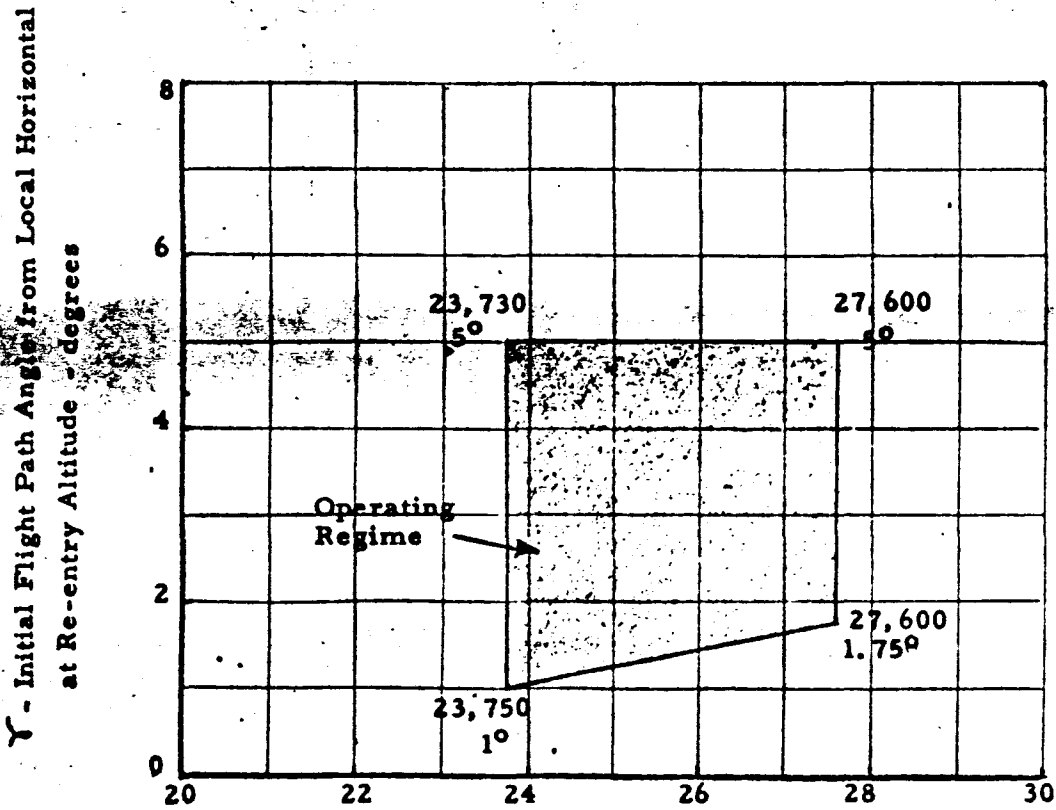
3.1.3.5 Safety

Normal transport, handling, and pad safety requirements will be recognized in the design and handling of Program vehicles, equipments 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.

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**Figure 7 - Operational Performance Envelope
Satellite Re-Entry Vehicles**



V - Earth Relative Velocity at Re-entry Altitude - ft/sec

Re-entry Altitude = 325,000 ft.

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3.2 System Design Control 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] and item hardware, shall be subject to review and approval by the 241 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. The Contractor shall state requirements for materials, parts and processes in his specification.

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 protected from 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.

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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 vehicle 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 IMSC 447969B. The payload shall conform to LMSC T3-6-002.

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. Due to security this requirement does not apply to payload equipments.

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 nominal weight for the SSO1B Agena for Program [REDACTED] missions shall be as follows for the conditions stated. These weights do not include the mission payload or SRV weights.

*For new or modified equipment procured subsequent to March 1965. MIL-STD-826 will supersede MIL-I-26600. The applicability of MIL-STD-826 and IMSC 447969 shall be in conformance with the requirements of the procurement contract.

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Nominal Agency Weight Summary

<u>Item</u>	<u>Weight (lbs)</u>	<u>Total Weight (lbs)</u>
<u>Weight Empty</u>		1899
Propellants	13560	
Helium	1	
Attitude Control Gas (-3 Mix)	68	
Aux. Alt. Control Gas - L/B (-3 Mix)	14	
5 IH Batteries	630	
5 DMU Rockets	100	
<u>Gross Weight Without Payloads</u>		16232
Less Adapter and Attach	354	
Less Retro Rockets	10	
Less Destruct System	6	
Less Horizon Sensor Fairings	7	
Less Attitude Control Gas	1	
<u>Ignition Weight w/o Payloads</u>		15854
Less Propellants	13560	
Less Engine Start Charge	1	
Less Attitude Control Gas	3	
<u>Burnout Weight</u>		2428
Less Residual Propellants	48	
Less Helium	1	
Propellant Margin	50	
<u>Weight on Orbit with Gas but w/o Payload</u>		2329
Less Remaining Att. Control Gas	64	
Less Remaining L/B Gas	14	
<u>Empty Weight on Orbit w/o Gas, w/o Payload, W 5 DMU</u>		2251
<u>Rockets & W/ 5 IH Batteries</u>		

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3.3.1.1.2 Satellite Vehicle Reliability Budget Requirements

The reliability design objective for the satellite vehicle is 0.97 for the period of time during which the vehicle functions as a second stage booster in the ascent mode, and 0.98 for subsequent operation in the active orbital mode for a period of 14 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 contractors shall prepare and maintain detailed reliability models 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 a receiver 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:

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- 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 by primary control and at least once by backup.
- G. Provide the necessary orbit maintenance capability (Drag Make-up System - DMS) when flying lower period and altitude orbits.

The Agena, SOIB, vehicle is a liquid-fueled second stage booster, powered by a gimbaled 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 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 adaptor 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 adaptor 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 adaptor 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 shall develop a nominal thrust of 16,000 pounds. The engine is designed for a nominal thrust duration of 245

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PROGRAM [REDACTED] VEHICLE

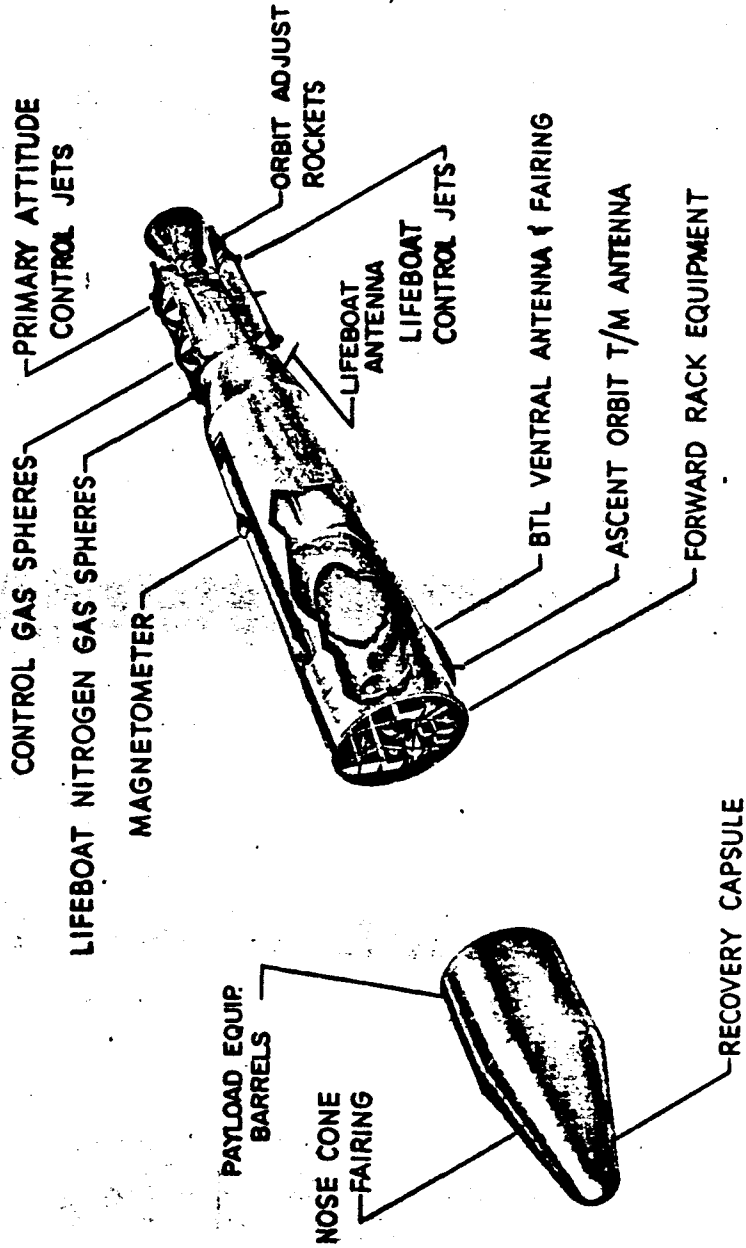


FIGURE 8 SATELLITE VEHICLE CONFIGURATION

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seconds. Propellants shall consist of unsymmetrical dimethylhydrazine (UDMH) fuel, and inhibited red fuming nitric acid (IRFNA) oxidizer. The propellant tanks shall be pressurized with helium to insure proper propellant pump operation and to maintain structural load carrying ability.

An electrical subsystem shall be provided to supply power to operate vehicle and payload electrical equipment. Batteries shall be used as the primary power source for the 24 volt direct current supply. Power conversion shall be accomplished by a three phase inverter(s) supplying single phase power from one leg, and DC to DC converters operating from the unregulated DC bus. Provision shall be made to supply switchable redundant three phase inverters.

The guidance and control subsystem shall sense vehicle attitude by means of horizon sensors and an inertial-reference gyro package. Pneumatic reaction-control jets shall 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 shall be supplied by main engine gimbaling activated by hydraulic servos. Vehicle velocity changes shall be 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 shall control the sequence of vehicle events. Steering of the vehicle during ascent shall be 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 shall consist of vehicle-borne transmitters, receivers, decoders, and programmers. Real-time commands shall be transmitted to the vehicle by PRELORT, ZEKE, 375 mc UHF (ZIPP) and Range Safety Command links. The PRECISION LONG Range Tracking radar (PRELORT) operating in the S-Band frequencies shall interrogate a vehicle-borne transponder for tracking. Commanding shall be accomplished for the digital (Zorro) and Analog Systems by modulation of the S-Band link. The Analog system employs pulse position modulation using combinations of six audio tones. Analog commands shall be used for selection of programmed payload and vehicle functions, and the Zorro commands to enable the primary re-entry sequence and the early A to B transfer.

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The VHF (ZEKE) command system shall employ a Very-High-Frequency (VHF) link where the carrier is amplitude modulated by audio tones. ZEKE commands provide the selection and execution of the "Lifeboat" back-up recovery sequence. A back-up command capability shall be supplied by a 375 mc UHF receiver in conjunction with a 39 command digital decoder. This system shall supply back-up commands to the existing analog decoder and shall also supply at least eight commands to the payload. Command Destruct (Thorad mounted) utilizes an Ultra-High-Frequency (UHF) link for range safety flight termination during vehicle ascent only until the time of Thorad 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 back-up information on payload status, diagnostic data and special payload 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 the additional transmitters and a four channel multicoupler.

An orbital programmer shall be 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 shall accommodate 13 brushes to make electrical contact with an externally grounded drum through the punched holes, thus providing a capability for 52 programmed commands. Tape speed shall nominally be 9 inches per subcycle and each tape length can be as long as 192 feet, providing programmed events for approximately 250 subcycles (orbit revolutions). Tape speed and positioning shall be adjustable by analog commands and back-up ZEKE commands to synchronize the programmer with the vehicle position and orbital period.

Two solid-state recovery timers shall be used to store commands for the recovery sequence through separation of the SRV from the satellite vehicle. The primary recovery timer shall be activated by a stored command from the orbital programmer, and the back-up recovery timer shall be activated by a ZEKE secure command to control the Lifeboat back-up recovery sequence.

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3.3.1.2.2 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.

Vehicle Systems Tests shall be conducted at the Contractor's plant prior to acceptance of the satellite vehicle by the procuring agency. The purpose of these tests 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 with accepted de-

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gradations in performance and reliability 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. Checkout Test Equipment
- E. Loading Equipment for Expendables
- F. Ground Electrical Power Equipment
- G. Ground Environmental Control Equipment
- H. Launch Control Equipment

3.3.1.2.3 Facilities

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:

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- A. Assembly Building
- B. Test Facilities
- C2. Clean Room

3.3.1.2.4 Satellite Vehicle Structural Envelope Requirements

Satellite vehicle equipment shall be contained within the structural envelope of the SO1B. 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.

3.3.1.2.5 Satellite Vehicle Effectiveness Requirements

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 equipments, together with logistical support of spares, shall be provided to support the launch and orbital objectives. Requirements for launch-holds and recycle from the launch pad shall not be more stringent for the satellite vehicle than for the payload.

3.3.1.3 Specific Design Requirements

3.3.1.3.1 Communication and Control Requirements (C & C)

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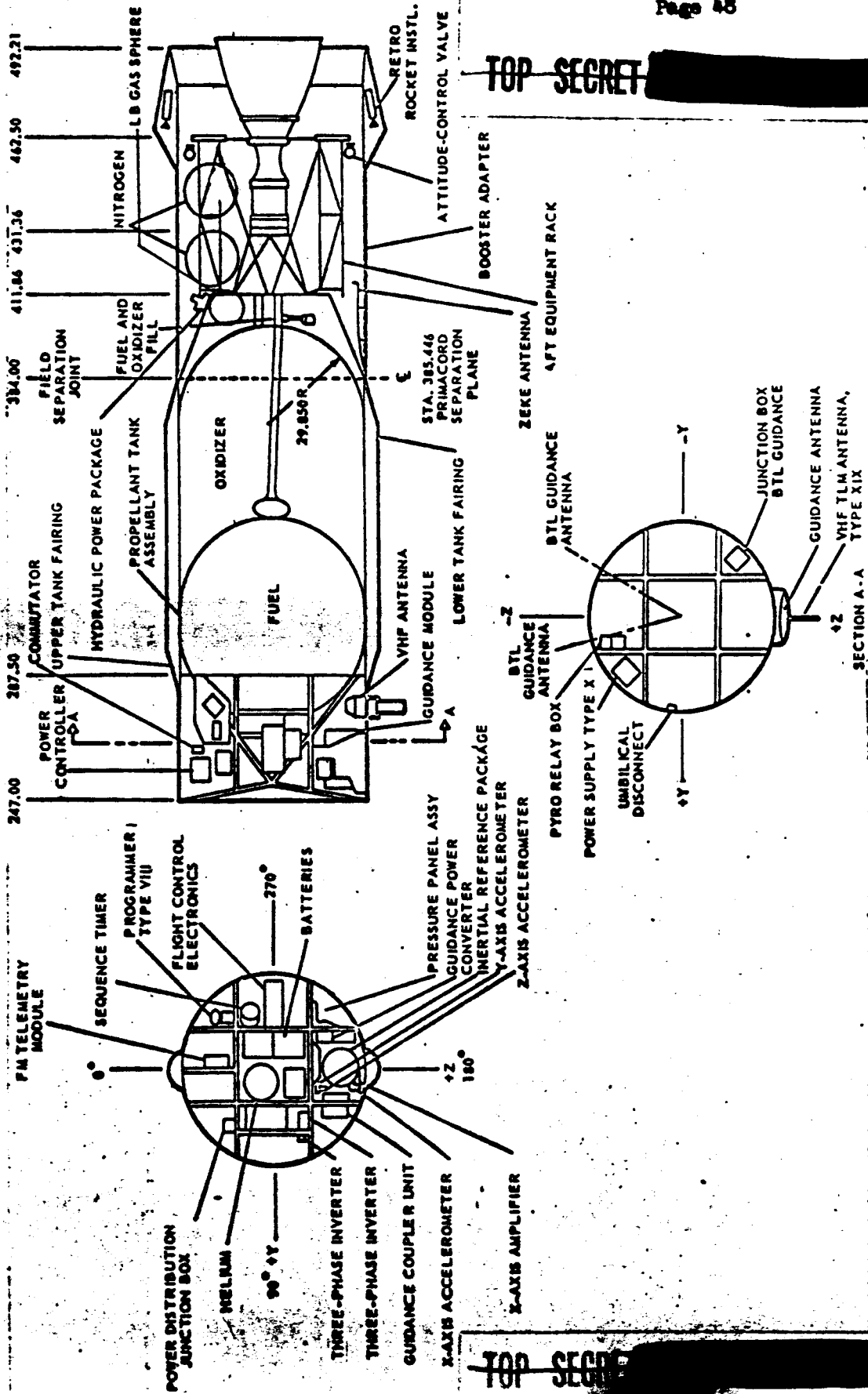


FIGURE 9 INBOARD PROFILE SATELLITE VEHICLE

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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 block diagram of the C & C subsystem is shown on Figure 10.

A. Real Time Commands

Each real time command shall be accompanied by functional telemetry verification (where possible) convenient in real time. Real time commands for Satellite Vehicle functions shall include but not be limited to those listed in Table 1.

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FIGURE 10
COMMAND SYSTEM BLOCK DIAGRAM

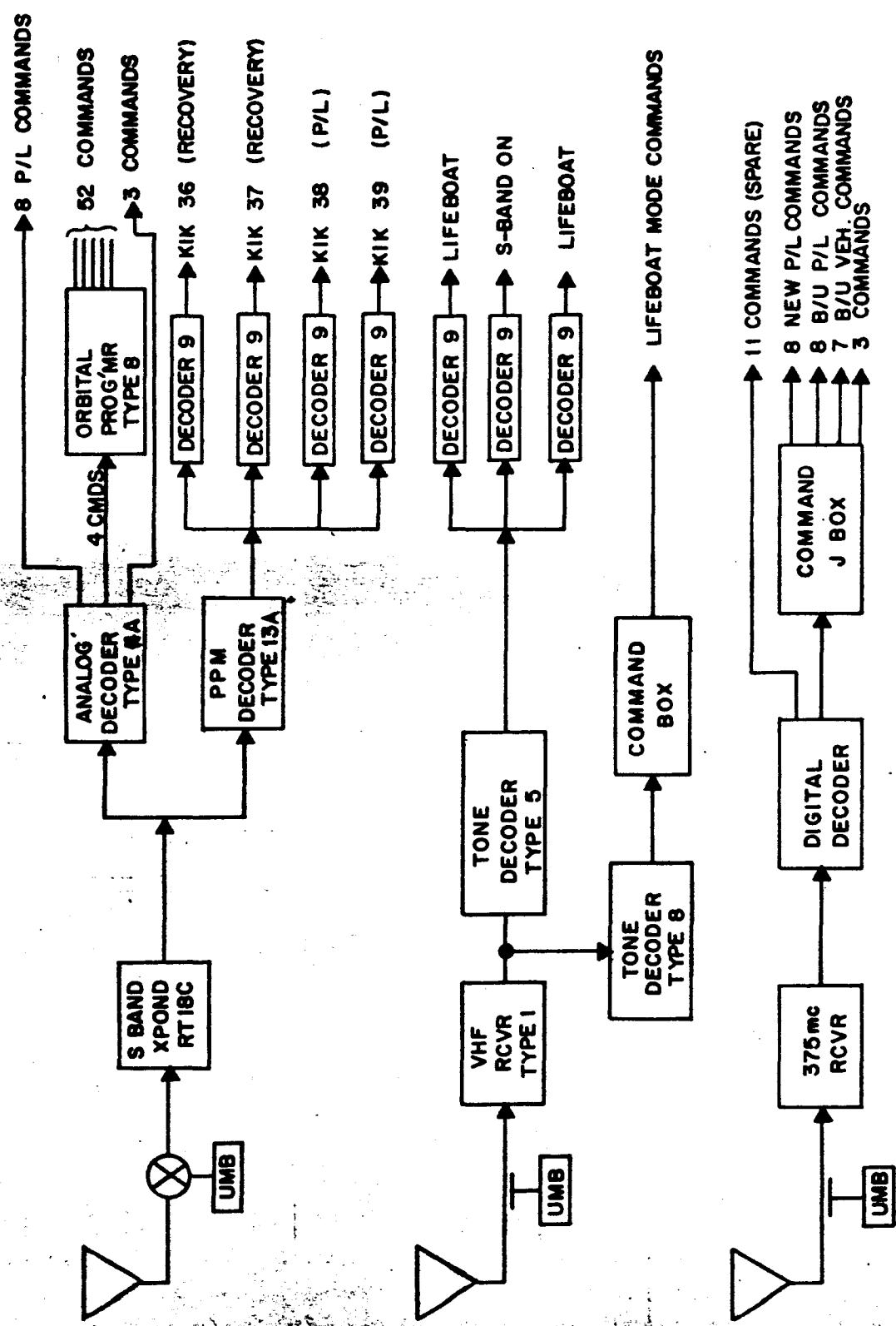


Table 1

	<u>Real Time Commands</u>				
	<u>Analog</u>	<u>Digital</u>		<u>Secure</u>	
	<u>S Band</u> <u>& No.</u>	<u>VHF</u>	<u>UHF</u>	<u>S Band</u>	<u>VHF</u> <u>S Band</u>
Orbital Programmer Increase/Decrease	1	X	X		
Orbital Programmer Ten Second Step	2		X		
Orbital Programmer Reset	3		X		
Select Even Orbit Recovery	4		X		
Select Odd Orbit Recovery	5		X		
V/h Start Level	6		X		
Orbital Programmer One Second Step	7		X		
V/h Half Cycle Level	8		X		
Camera Program Select	9		X		
V/h Delay Start Position	10		X		
Panoramic Camera Mode Select	11		X		
Operation Selector No. 1	12		X		
Drag Make-up System Enable	13		X		
DISIC Camera Mode Select	14		X		
Operation Selector No. 2	15		X		
Lifeboat Next Orbit		X			
Primary Next Orbit		X			
Panoramic Camera No. 1 Exposure Control Fail Safe				X	
Panoramic Camera No. 2 Exposure Control Fail Safe				X	
Panoramic Camera No. 1 Filter Change			X		
Panoramic Camera No. 2 Filter Change			X		
Exposure Control Delay			X		
Yaw Programmer Enable/Disable			X		
DISIC Camera East/West/Both/Off			X		
Emergency Ops. Select/Mode Select Bypass			X		
Fan Early A to B Switchover					X
DISIC Early A to B Switchover					X
Recovery Enable #1					X
Recovery Enable #2					X
Lifeboat Enable #1					X
Lifeboat Enable #2					X
Telemetry & Beacon On (for 420 seconds)					X

Radio Guidance Commands

The radio guidance command link shall use a continuous tracking X-Band radar which shall pulse-position modulate the command spacing between continuous pairs of address pulses. The commands used are listed in Table 2.

Command Destruct

The Stage 1 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 and payload functions shall include but not be limited to those listed in Table 3.

Table 2

Radio Guidance Commands

<u>Discrettes</u>	
Sequence 0	No command
" 1	Main Engine CutOff (MECO) - Stage 1
" 2	Command Separation
" 3	Stage 2 Velocity Meter Enable.
<u>Commands</u>	
Pitch Up Steering Command	Booster and Agena
Pitch Down Steering Command	" " "
Yaw Right Steering Command	" " "
Yaw Left Steering Command	" " "

Table 3

STORED COMMANDS - VEHICLE

<u>Function</u>	<u>Number of Commands</u>	<u>Command Mode</u>
<u>Ascent Phase</u>		
Booster Separation Backup	1	Standard Timer
Enable Radio Guid. Steering	2	" "
Engine Ignition Sequence	4	" "
Flt. Control & Attitude Functions	15	" "
Disable Radio Guid. Steering	3	" "
Engine Shutdown Sequence (Backup)	4	" "
Switch TM to Orbit Mode	2	" "
Switch Flt. Control to Orbit Mode	1	" "
Arm Backup Rev. Pyro Pwr.	1	" "
Enable Primary Rec. Select	1	" "
Stop Ascent Timer	1	" "
<u>Orbit Phase</u>		
Beacon & TM Link I on	1 Brush	Orbital Prog. ←
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 "	" "
Re-entry Execute-even Orbit	2 "	" "
TM Link II On	1 "	" "
TM Link II Off	1 "	" "
Pre-launch Tape Index	1 "	" "
Drag-Make-up	3 "	" "
<u>Recovery Phase - Primary</u>		
Vehicle Pitch-down Maneuver	4	Primary Recovery Timer (DI/AN)
Flt. Control & Attitude Functions	3	" "
Vehicle Pitch-up Maneuver	2	" "
Reset Recovery Timer	1	" "
Remove Rec. Timer Power	1	" "
Payload Commands (See Table 4)	4	" "
Recovery Phase - Lifeboat		Lifeboat Recovery Timer (DI/AN)

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Table 4

STORED COMMANDS - PAYLOAD

<u>Function</u>	<u>No Commands</u>	<u>Command Mode</u>
<u>Ascent Phase</u>		
Switch Power for Early Door Eject	1	Sep. Switches
Switch Power for Camera, Cassettes and Door Eject	1	Standard Timer
<u>Orbit Phase</u>		
Orbital Counter	1	Brush(s) Orbital Prog.
Clock Interrogate	1	" " "
V/h Programmer Start Pulse	1	" " "
T/M Enable	1	" " "
Camera (ON_OFF) Sequence Programs 1 thru 9	18	" " "
Redundant Off for All Prog	1	" " "
Tape Recorder Read-in	1	" " "
Tape Recorder Read-out	1	" " "
Tape Recorder Stop	1	" " "
Tape Recorder Track Switch	2	" " "
Payload Exposure Control	3	" " "
DISIC ON/OFF	2	" " "
Brush Intermix Advance	1	
<u>Recovery Phase</u>		
Both Primary and Lifeboat		
Arm No. 1 Capsule Eject	1	Recovery Timer (DI/AN)
Transfer Pwr. to No. 1 Capsule	1	" " "
Electrical Disconnect	1	" " "
Fire No. 1 Capsule Eject Squibs	1	" " "
Switch Over to B Mission	1	P/L Function
Arm No. 2 Capsule Eject	1	Recovery Timer
Transfer Pwr to No. 2 Capsule	1	" "
Electrical Disconnect	1	" "
Fire No. 2 Capsule Eject Squibs	1	" "

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1. Standard Timer

The Standard timer shall provide capability for discrete time interval commands with two 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. Orbital Programmer

The orbital programmer shall provide a capability for 52 stored commands (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 maintaining synchronization by resetting \pm 150 seconds of programmed events with the determined position of the Satellite Vehicle. Accuracy of the orbital programmer shall be plus or minus 3.5 seconds, including the effect of tolerances on tape punching.

3. Recovery and Lifeboat Timers

The recovery and Lifeboat timers shall provide a capability to initiate at least 14 events each. 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 timer at the time of its last event. Timer accuracy when installed in a system shall be plus or minus 0.5 seconds or 0.1 percent between events, whichever is greater.

3.3.1.3.1.2 Instrumentation and Telemetry

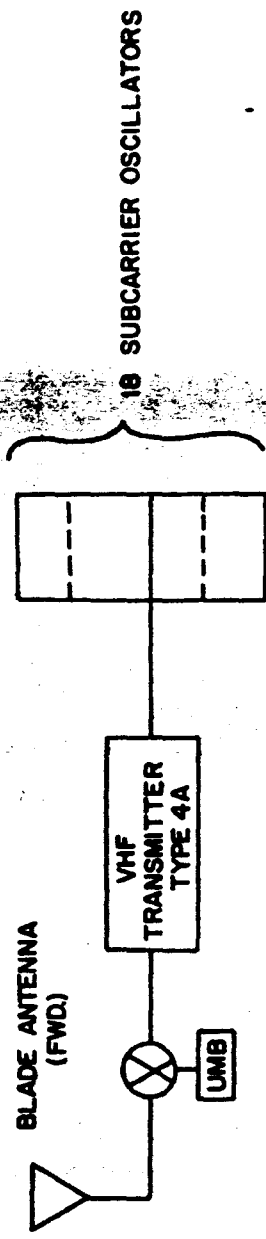
Provisions shall be made to incorporate instrumentation in the Satellite Vehicle to provide timely and accurate data for the pre-launch, launch, orbital and the recovery separation phases of operation. Considerations shall be given to data requirements on a real-time and post-flight basis. The Satellite Vehicle shall provide the telemetry equipment for transmitting vehicle and payload real-time and stored data to the ground stations. FM/FM transmissions in the 215 to 260 MC/sec band shall be employed. (Figure 11)

A. Sensors

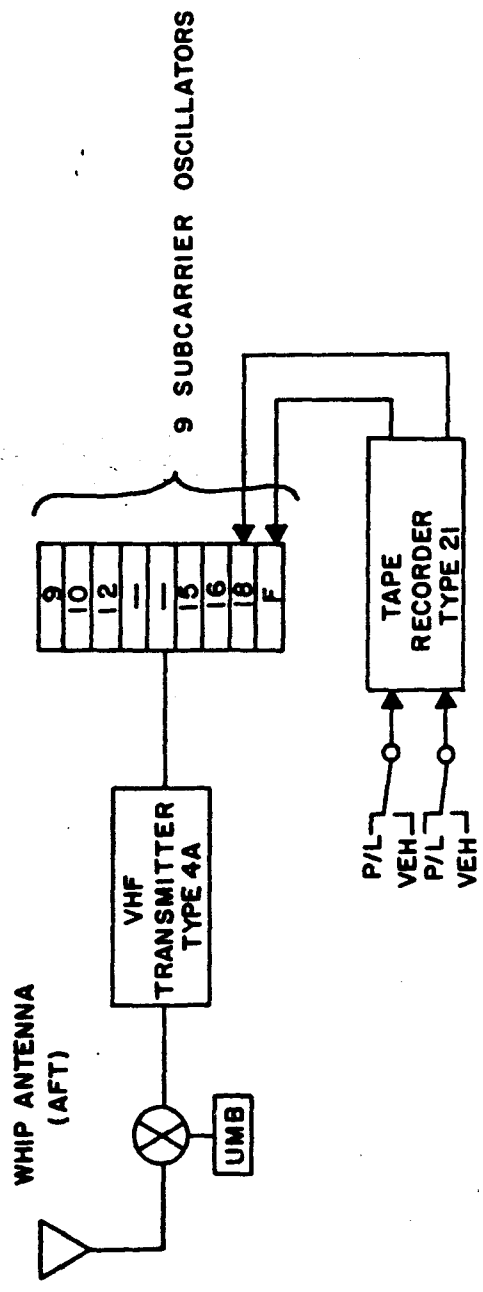
Sensors shall be provided as required for instrumentation of the satellite vehicle. Sensors shall have adequate dynamic range, frequency response characteristics and accuracy to meet the program requirements. Insofar as possible, sensors shall provide an output that can be directly correlated with the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

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FIGURE 11. TELEMETRY SYSTEM BLOCK DIAGRAM



LINK 1 (TYPE 5)



LINK 2 (TYPE 7)

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B. Reserved Subcarriers for Payload

The payload will require 10 channels, four of which shall be commutated, two on a 0.4 x 60 commutator (24 samples-per-second) and two on a 5 x 60 commutator (300 samples-per-second). Critical monitors shall be supplied on more than one T/M channel.

C. 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 $\pm 3\%$.

Maximum overall error for commutated data, which has gone through the cycle of vehicle tape recording and subsequent playback, shall not exceed $\pm 5\%$.

Under conditions of RF signal strength well above threshold, no typical data channel shall contain hum, ripple or noise with a combined amplitude exceeding a 2% RMS value with respect to a full scale data range.

For telemetry signal reception and processing, receivers with an IF bandwidth of 300 KC/sec shall be used. Subcarrier discriminators shall utilize IRIG 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 MHz 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 $\pm 2\%$ of the bandwidth through which the subcarrier's frequency is deviated by full scale data. The subcarriers frequency deviation shall be

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proportional to the modulating voltage (positive frequency deviation for a positive modulating voltage) with a linearity within $\pm 0.5\%$. Harmonic distortion shall not exceed $\pm 1\%$. As a design objective, no subcarrier oscillator shall be capable, under malfunction conditions, of generating an output frequency which interferes with other subcarrier oscillators.

F. Commutators

Commutators shall have proven reliability and compatibility with SCF decommutation equipment. Either non-return-to-zero or return-to-zero pulse train formats may be used. The total error contribution of any commutator for all combined causes shall not exceed $\pm 1\%$ of full scale. At least three calibration points shall be provided in each commutator's pulse train which are directly referenced to the applicable instrumentation points.

G. Commutated Data Grouping

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

H. 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.

I. 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, or in the case of the payload, data may be expressed as a function of voltage.

The Agena SOIB 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.

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Table 5. Typical Assignment of Data Channels

Channel No.	Mode	
	Ascent	Orbit
Link 1		
1 - 2	Vehicle	Vehicle
3	Vehicle	Vehicle
4	Vehicle	Vehicle
5	Payload	Vehicle
6	Vehicle	Payload
7	Vehicle	Payload
8	Vehicle	Payload
9	Vehicle	Payload
10	Vehicle	Payload
11	Vehicle	Payload Commutated (0.4 x 60)
12	Vehicle Commutated	Vehicle Commutated
13	Payload	Payload Commutated (0.4 x 60)
14	Vehicle	Vehicle
15	Vehicle Commutated	Vehicle Commutated
16	Vehicle Commutated	Vehicle Commutated
17	Vehicle Commutated	Vehicle Commutated
18	Vehicle	Payload
Link 2		
9	Dig. Dec. (ZORRO)	Dig. Dec. (ZORRO)
10		
12	Vehicle B/U Commutator	Vehicle Back-up Commutator
15	P/L Commutated (5 x 60)	Payload Commutated (5 x 60)
16	P/L Commutated (5 x 60)	Payload Commutated (5 x 60)
18	Payload	Tape Recorder Track 1
P	Payload	Tape Recorder Track 2

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Telemetry subsystem specification and the telemetry channel assignment lists for each flight shall require approval of the [REDACTED] Program Directorate and the PSAPO 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.

3.3.1.3.2 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 SOLB equipment shall conform to the requirements of LMBC 6117B (or D), "General Environmental Specification for the Agena Satellite Program" except as specifically authorized. All P/L equipments shall conform to the requirements of T3-6-002.

3.3.1.3.3 Guidance and Attitude Control Requirements

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

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 de-boost

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and OAS phases within the bounds of practicability and reliability. It shall be a system requirement to back-up the critical de-boost sequences with a redundant attitude control and orientation system capable of one operation minimum.

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 discrete command, the satellite vehicle standard 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 the optical doors shall be ejected and the control of the satellite vehicle attitude and rates about all three axes shall be initiated, utilizing the vehicle reference attained at VECC. The horizon sensors shall reference the roll axis to the earth horizon. Engine ignition shall be initiated by signal from the standard timer and engine shut-down initiated by the velocity meter after a predetermined velocity to-be-gained has been achieved (backed up by a standard timer signal). The velocity meter shall be enabled by a radio guidance discrete command. During the burn period, pitch and yaw control shall be provided by hydraulic actuation of the gimballed engine while roll control shall be maintained by pneumatic reaction control jets. Radio guidance commanding shall be utilized throughout 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 missions are as given in Table 6.

Table 6 Dispersions at Orbit Injection

Parameter	Requirement (3 σ)	Objective (3 σ)
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° (e > .008) \pm 180° (e \leq .008)	\pm 5° (e > .008) \pm 45° (e \leq .008)
Inclination Angle	\pm 0.25°	\pm 0.25°

* Note: With propellant contingency in Stage II to insure 90% probability of success to achieve period requirement.

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B. Orbital Phase - Throughout the orbital mission, excluding de-boost sequence and orbit adjust maneuvers the satellite vehicle shall remain oriented in the "nose first" position with the roll axis of the vehicle aligned with the resultant ground track velocity vector and normal to an 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 momentum unbalances:

<u>Axis</u>	<u>Torque</u>	<u>Worst Case Mode</u>
Pitch	1.09 ft lb seconds	Mono
Yaw	4.04 ft.lb.seconds	Stereo *
Roll	6.45 ft.lb.seconds	Mono

The maximum restoring torque capability of the Guidance and Control system is:

<u>Axis</u>	<u>Torque</u>
Pitch	8 ft.lb.
Yaw	16 ft.lb.
Roll	2 ft. lb.

Pointing accuracy requirements and maximum limit cycle rates are given in Table 7.

Table 7 Pointing Accuracy and Rates

Function	Requirement	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

* Mono value is one-half the indicated max. yaw momentum.

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The above accuracies are required during payload operation. At times of no payload operation, reduced accuracies are acceptable. In addition to the above requirements for stabilization, the satellite vehicle shall be capable of being maneuvered in yaw. In response to a payload yaw programmer voltage, the vehicle shall be positioned in yaw to 0.25 degrees per 1.67 millivolt of roll torque input with a stabilization time of 6 minutes.

C. De-Boost Phase

The de-boost sequence for the satellite vehicle is controlled by signals from a DI/AN recovery timer. Upon the programmed command, the satellite vehicle shall pitch down a nominal 120 degrees from the local horizontal, while orbiting in the "nose-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 be approximately 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 (37)	Objective (35)
Pitch Angle from Local Horizontal	$\pm 7.9^\circ$	$\pm 6.5^\circ$
Yaw Angle from Orbit Plane	$\pm 1.5^\circ$	$\pm 1.1^\circ$
Roll Angle from Radius Vector	$\pm 1.3^\circ$	$\pm 1.0^\circ$

In the event of a malfunction in the primary attitude control subsystem, a back-up stabilization system (Lifeboat) shall be activated by a secure real-time command. This Lifeboat system shall be capable of performing all de-boost sequences necessary for properly ejecting one re-entry vehicle from the satellite vehicle. Upon initiation, the Lifeboat subsystem shall be capable of orienting the satellite vehicle from a tumbling mode of 20 degrees per second about any axis, and be capable of holding the de-boost orientation for a minimum of 30 seconds. Lifeboat attitude control is established by lining up the vehicle roll axis with the local magnetic vector.

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and keeping the roll rate below ± 2 degrees/second. Lifeboat shall be capable of acceptable performance on North to South passes, and the ability to perform acceptably on South to North passes is desirable but not presently required. Pointing accuracy required for Lifeboat is plus or minus 10.5 degrees to the local magnetic vector referenced to the required 120 degree pitch down orientation. The vehicle shall not be re-oriented after Lifeboat usage.

D. Drag Make-up Phase

The vehicle shall have the capability to fire in boost or de-boost by using one of a set of solid rocket motors. These rockets have the following characteristics:

Vacuum Thrust	137 pounds
Total Impulse - Vacuum (Burn Times = 16.5 sec)	2250 pound-seconds
Nominal Delta Velocity/Rocket	17 feet/second

E. Satellite Vehicle Mass Characteristics

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

Table 9 - Estimated Mass Properties, Satellite Vehicle

Condition *	Wt. Lbs.	Center of Gravity Station (inches)			Moment of Inertia (Slug ft ²)		
		X	Y	Z	I _y (Pitch)	I _x (Yaw)	I _z (Roll)
Separation from Stage I Booster	17601	327.1	0.20	0.14	16,454	16,437	312
Ignition Weight	17593	327.2	0.20	0.13	16,481	16,464	310
Burnout Weight	4116	270.1	0.86	0.56	9,977	9,960	309
Wt. On-Orbit	4017	270.1	0.86	0.56	8,867	9,850	309

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 direct current power source shall consist of batteries having sufficient capacity to supply electrical energy for all

* NOTE: Does not include 5DMU rockets

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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 operation. The power source shall maintain its voltage between the limits of 22 and 29.5 volts D.C. measured at the vehicle bus. A capability to provide electrical energy for mission durations of up to 14 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. Electrical power profile characteristics including surge characteristics of pyrotechnic devices and payload equipment.
- d. Activated stand time capabilities of the battery versus time from battery activation through mission completion.
- e. Battery weight as a function of mission duration and allowable weight and space versus performance capability for the ascent flight.
- f. A power margin of 5 percent over and above the predicted -3σ 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 (T3-5-023). Estimated maximum payload power requirement is 900 watt hours per day with the following requirements: (14 day average)

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a. Unregulated DC with a maximum load of 30 amps continuous, with 60 amp peak not to exceed 500 milliseconds (duty cycle of 20 minutes on and 70 minutes off).

b. 400 \pm .008 cycles at 115 \pm 2% volts AC with an average load on Phase C of 0.6 amps continuous and peaks of 0.85 amps for periods not to exceed 500 ms (with a duty cycle of 20 minutes on and 70 minutes off); and Phase A with an average load of .1 amps continuous.

c. Unregulated voltage for pyrotechnic actuation with peak currents of 60 amperes and peak duration of 5 milliseconds while maintaining a buss voltage of 14 to 29.5 volts D.C. at the payload pyro connector.

C. Electromagnetic Interference Control

The satellite vehicle should be designed to comply with MIL-E-6051C. All subsystems of the satellite vehicle should comply with MIL-I-26600 and MIL-E-6051C except as specifically authorized: components should comply with IMSC 447963B (MIL-I-26600). An electrical interference control plan should be originated and employed in the design, fabrication and test of the satellite vehicle. The payload shall conform to Specification IMSC T3-6-002.

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 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 (or payload interface), and return, attributable to the harness, including connectors, shall not exceed 0.5 volts D.C. Voltage drops in primary leads of up to 1 volt are permissible 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.

A All ascent pyrotechnic loads shall be connected to a separate, diode-isolated primary battery. The isolation diode may be shorted out at the completion of the ascent phase of flight. Fusistors shall be provided

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in all pyrotechnic circuits to protect the vehicle power source and distribution networks from short circuits that may occur during and after pyrotechnics firing. Wiring circuits to pyrotechnics and return shall be capable of handling the maximum all-fire current of the pyrotechnic device. Power and pyrotechnic harnesses may be grouped and routed together, but shall be separated (as an objective) from harnesses for instrumentation, commands, and test plugs.

E. External/Internal Power Transfer Switch

An external/internal power transfer switch shall be used before liftoff to transfer the AGE power supply to the satellite vehicle operational power supply. The transition from external to internal power 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 IMSC 447969B (Agena).

G. Interface Connectors

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

- a. Pyrotechnics
- b. Electrical Power
- c. Commands
- d. Telemetry Instrumentation
- e. Test

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.

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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 operations by means of yaw and pitch thrust chamber movement. Cylindrical shaped propellant tanks shall be provided to contain a nominal propellant load of 13,520 lbs. Propellants shall consist of Inhibited Red Fuming Nitric Acid (IRFNA) as an oxidizer, the Unsymmetrical Dimethylhydrazine (UDMH) as a fuel.

For the [REDACTED] Mission, a single propulsive interval is required for the satellite vehicles, and restart of the rocket engine shall 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 RSS dispersions in 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 241 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, unless directed to the contrary by the [REDACTED] Program Directorate.

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3.3.1.3.7 Guidance Requirements - Ascent

The satellite vehicle shall contain the vehicle-borne radio guidance group for 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 SLV-2G/SOLB Electrical Interface Control Specification MSF 1418956 and the Electrical Interface drawing 1376862.

The function of the radio guidance system shall be to increase guidance accuracy by providing real-time sequenced events and real-time steering corrections to the Thorad and Agena vehicles during their boost phases. Steering corrections, implemented by the radio guidance system 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 a pre-programmed mode. Similarly, the sequenced events for separation of Stage I from the satellite vehicle, and start of the satellite vehicle Standard Timer shall be actuated by programmed stored commands if they are not commanded by radio guidance. It is not necessary that Thorad MECO and Agena velocity meter enable commands be backed-up by a programmed command, (since engine shut-downs will occur upon propellant depletion).

Accuracy requirements for 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 satellite vehicle 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. UHF command destruct 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 a 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.

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The SOLB flight termination subsystem shall be equipped to meet the minimum command destruct requirements of the Western Test Range as specified in Air Force Western Test Range Manual 127-1. Command Destruct signal provisions shall be in accordance with the Interface Control Specification between SLV-2G/SOLB (IMBC 1418956) and the Electrical Interface Drawing 1376862.

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3.3.2 Payload

3.3.2.1 Payload General Design Requirements

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 (station 247) as shown in Figure 1. Total Payload weight shall be less than 1750 lbs. The primary camera equipment shall consist of two hi-acuity panoramic cameras mounted in a 30 degree convergent stereoscopic configuration. Simultaneous operation of both cameras provides stereoscopic photography. During camera operation, the lens assembly rotates (through 360 degrees) about its vacuum nodal point to perform the pan function. As the exposure slit in the scan head traverses the main film frame, horizon-sighting cameras, one located on each end of the camera film track, shall record a vehicle attitude reference photograph on the film between panoramic frames. Time reference data and identifying information shall be recorded on the edge of the pan frame. Lens rotation and film transport shall be powered from the camera electric drive-motor through a system of cams, belts and gears. Camera cycle rate and the DMC programmer voltage shall be directly proportional, selectable by commanding the DMC programmer, to match the V/H condition. The camera system shall be capable of operation between the altitudes of 80 and 200 n.m. with performance optimized in the area between 80 and 120 n.m.

Film tension is provided on each side of the panoramic cameras by drive motors in the supply and take-up cassettes. Film from both pan cameras shall be routed through the "B" satellite recovery vehicle (SRV), whose take-up spools are locked to prevent rotation during the "A" mission, to the "A" SRV. Upon command, the film entering the "A" SRV shall be cut, the spools in the "B" SRV unlocked, and take-up initiated for the "B" mission.

A Dual Improved Stellar Index camera (DISIC) shall be installed adjacent to the "B" SRV to provide terrain and stellar photography for "A" and "B" missions.

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

At the conclusion of the "A" mission, the "A" SRV shall be 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 "A" and "B" SRV's shall be retained on the Satellite Vehicle until jettisoned by a signal in the deboost sequence for the second SRV. The satellite recovery vehicle (SRV) shall be a modified Mark 5A recovery sub-system described in Section 3.3.3.

Aerospace Ground Equipment (AGE)

The camera equipment and satellite recovery vehicles shall be provided as Government Furnished Equipment (GFE) to the Payload Contractor for integration into the mission configuration. The flight Payload shall be acceptance tested at the Payload Contractor's facility prior to shipping to the launch base in a flight-ready condition (P/L factory-to-launch).

At the launch site, the Payload shall be inspected to ensure that no damage has been sustained as a result of shipment. Inspection shall not require disassembly. Functional and compatibility tests shall be performed between the Payload and the Satellite Vehicle during the Payload mating operation.

Mechanical and electrical AGE shall be provided at the Contractors' facilities and at the launch base to implement the Factory-to-Launch 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.

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

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related to the [REDACTED] Program shall be identified and substantiated. Facilities shall be approved by the PSAPO. Facility requirements shall consider, but not be limited to, the following:

- A. Assembly Building
- B. Test and Checkout Facilities
- C. Clean Rooms

Payload Structural Envelope Requirements

Payload equipment shall be contained within the structural envelope of the payload section of the Satellite Vehicle. Figure 12 presents the general arrangement of primary payload equipment and major dimensions. The maximum diameter of the Payload 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.

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 launch, and from lift-off through separation of SRV's. The satellite re-entry vehicles shall each contain beacon and telemetry subsystems for operation during the re-entry/recovery phases as described in Section 3.3.3.

Payload Commands

The Payload shall be capable of programmed operation for any portion of the ground track on any orbit during the active mission life. Pre-programmed stored-commands shall be capable of adjustments through real-time commands to match the payload periods of operation and photographic

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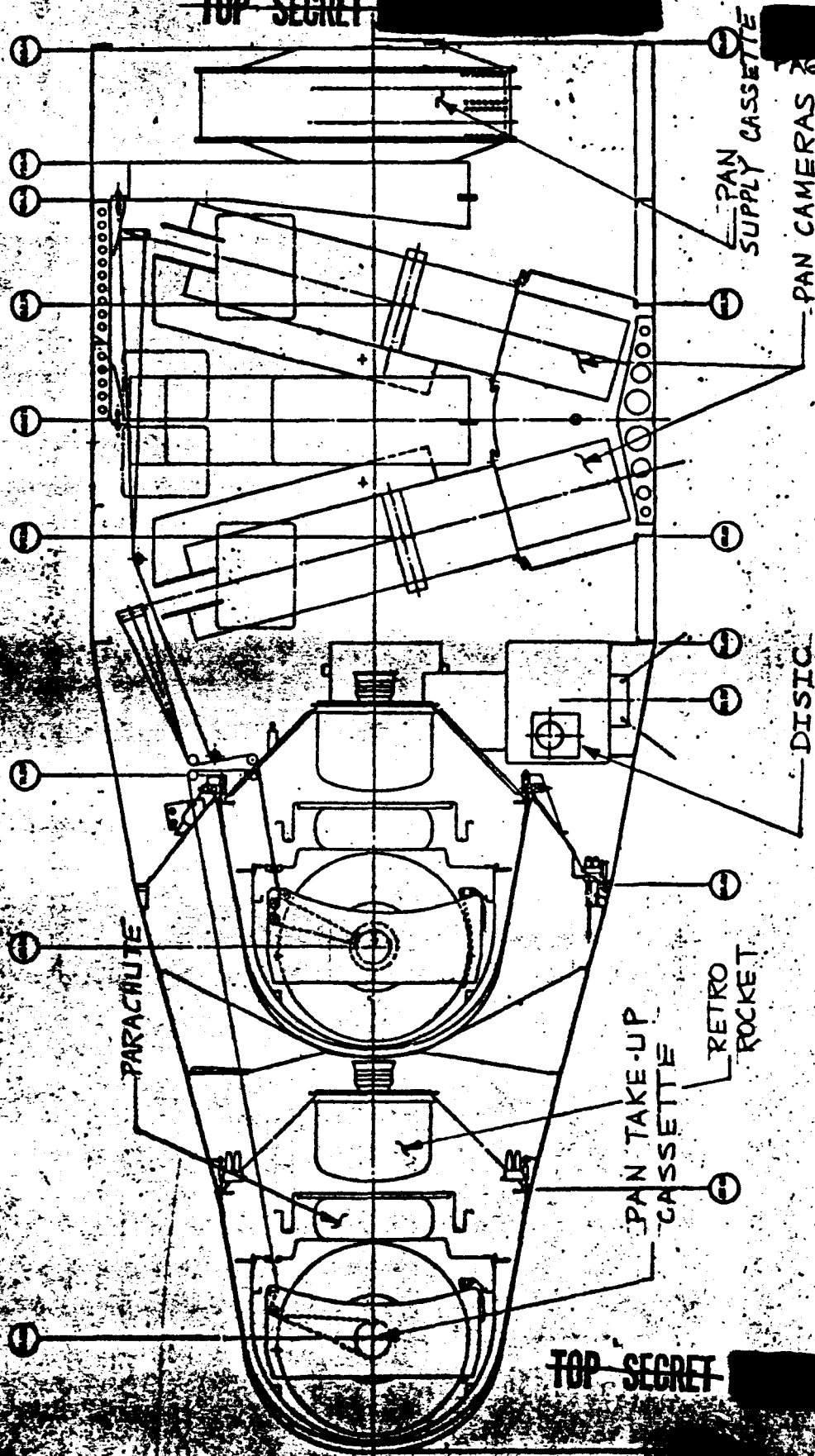


FIGURE 12 PAYLOAD INBOARD PROFILE

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equipment functions to actual orbital and ground track conditions during the mission. Command functions shall provide adequate flexibility to accommodate the combined effects of ascent flight dispersions (10°), and variations in orbital parameters due to atmospheric drag and earth oblateness. Payload real-time commands shall be accompanied by a functional telemetry verification (where possible) in real-time. Real-time and stored command requirements for the payload shall be defined in detail in the Satellite Vehicle/Payload Electrical Interface Specification (IMSC T3-5-023).

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. Telemetered data shall provide information adequate for determining equipment mode status and state-of-health, and shall constitute a basis for real-time command decisions and malfunction diagnosis.

The telemetry links for payload data shall be provided by the Satellite Vehicle by means of the FM/FM transmitters located on the vehicle side of the interface. The Satellite Vehicle shall also provide a tape recorder to accommodate storage of payload data with subsequent playback over SCF ground stations. Payload instrumentation signals shall be suitably conditioned for compatibility with telemetry equipment in the Satellite Vehicle. Requirements for payload telemetry shall be defined in detail by the Satellite Vehicle/Payload Electrical Interface Specification (IMSC T3-5-023).

Equipment Environmental Requirements

The J3 System shall orbit the earth in a vacuum environment and under conditions of solar radiation varying from direct sunlight to earth shadow.

The Payload Integrating Structure shall provide the necessary mountings for payload equipment. Mounting provisions shall be capable of achieving the high degrees of alignment accuracy required for camera optics

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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 control 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 allight proof environment, free of all light leakage that could produce objectionable film fogging.
- B. A pressure environment shall be provided to suppress corona discharge. The pressure makeup unit (PMU) shall be capable of maintaining pressures of 20 microns or higher in the payload cylindrical sections during camera operation.
- C. Detachable doors shall be provided in the payload structure for optical viewing ports. These doors shall provide protection to optical equipment during ascent and 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. Passive thermal control shall be provided (where possible) for temperature-critical equipment. The external surface of the payload structure shall be used for passive temperature control. An optimum absorptivity shall be provided by surface coatings and mosaic patterns to maintain an average temperature of 70 ± 30 degrees Fahrenheit inside the cylindrical section of the payload.

Power Supply Requirements

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. Payload electrical power requirements shall be specified in detail in the Satellite Vehicle/Payload Electrical Interface Specification, (LMSC T3-5-023). The payload section shall provide power distribution networks for all payload equipment forward of the

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Satellite Vehicle/Payload interface (Station 247). This shall include all junction-boxes, cables and connectors necessary for the control and monitoring of payload equipment.

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 T3-6-002.

Design Requirements - Payload Equipment

The following is a generalized description of the equipment located in the J-3 Payload section. For detailed description of this equipment, refer to the applicable paragraphs of "J-3 Payload Section-Corona J Program" (HO-J3-001).

A. Panoramic Camera

The panoramic cameras shall be capable of generating stereoscopic or monoscopic photographs at altitudes between 80 and 200 n.m. Film width shall be 70 millimeters. The cameras shall have a sustained operational capability of 20 minutes operation per single orbit. The panoramic scan angle shall be 70 degrees with image motion compensation provided during scan exposure. Data required to identify mission reference time and camera serial number shall be recorded during exposure of panoramic frames. Start of pass shall be recorded at the beginning of each instrument startup. Corona marking shall be limited to the first five consecutive frames from the start of pass mark at each instrument startup.

B. Performance

The panoramic camera lens shall be a Petzval with an aperture of $f/3.5$ and a focal length of 2 1/2 inches. The lens shall be focused for orbital vacuum conditions. Each panoramic camera shall demonstrate a minimum dynamic resolution of 120 lines per millimeter using a USAF standard (2.1) test target and 100 percent match of image motion compensation.

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Image Motion Compensation

Each panoramic camera shall incorporate an Image Motion Compensation (IMC) mechanism to provide relative velocity correction during the scan exposure. The IMC shall be accomplished by a cam that nods the instrument at a velocity proportional to the camera cycle rate. The range of IMC adjustment shall accommodate camera operation at orbital altitudes from 80 to 200 nautical miles with orbit eccentricities ranging from zero to 0.033. The camera cycle period shall be adjustable by command in flight and shall be repeatable to 1 percent.

Scan Drive

The scanning drive mechanism shall drive the lens cell over the 70 degree panoramic scan angle. Velocity of scan shall be controlled so as to produce no objectionable banding in ground scenes. Installation of the cameras shall provide counter-rotation of moving components during operation of both cameras. To eliminate light leakage through the optical train during extended periods when photographic operations are not required, provisions shall be made to orient both of the lens cells in a "homed" position such that lens elements are not exposed to the ambient light sources.

Camera Data Readout

A Silicon light pulser data head shall be used for each 70 mm panoramic camera to record system time. The camera serial number, start of pass indicator and 200 pps timing track shall also be recorded on each frame and shall not interfere with panoramic or horizon optics photography.

Horizon Cameras

Two 55 millimeter focal length, $f/6.3$ 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.

C. Digital Recording Clock Generator (DRCG)

The Digital clock shall be capable of storing time unambiguously

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for a period of 6.2 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 and DISIC film.

D. DISIC Subsystem

The subsystem shall consist of one DISIC camera with two take-up cassettes, one supply cassette, exit housing, film chutes, baffles and 2000 feet of terrain film and 2000 feet of stellar film. The DISIC subsystem shall have the capability of independent programmed operation or concurrent operation with the panoramic system. Exposed film shall be transported to cassettes within the recovery systems via a film path independent from the panoramic film path. The "A" and "B" film paths shall contain back-up film cutters. The DISIC system shall be fused to protect the panoramic system from a DISIC subsystem power failure. DMS shall furnish a Control Sequence box to provide control of the DISIC subsystem. DMS shall provide a Cut & Splice mechanism (TUNA) to accomplish the transfer of the take-up operation from the "A" to the "B" DISIC take-up cassettes.

Both the DISIC and panoramic cameras' take-up cassettes shall be mounted within the Satellite Recovery vehicles. The "B" mission spools, installed in the "B" SRV, shall be held to a fixed position until programmed to initiate take-up.

3.3.3 Satellite Recovery Vehicle

3.3.3.1 SRV General Design Requirements

3.3.3.1.1 General Description

The basic recovery subsystem consists of two Mark 5A Satellite Recovery Vehicles mounted in tandem on the payload structure as shown on Figure 12. 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

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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 Contractor for installation of the take-up cassettes and parachutes. Attachment and separation provisions, including command signals and instrumentation, shall be provided by the Payload Contractor. Figure 13 illustrates the general arrangement and primary components of the basic SRV.

3.3.3.1.2 Aerospace Ground Equipment (AGE)

Existing AGE will be used to extent practicable.

3.3.3.1.3 Facilities

No special facilities other than clean rooms are expected to be required for the SRV. Any requirements of this type, uniquely related to Program 241, shall be identified and substantiated.

3.3.3.2 Specific Design Requirements

3.3.3.2.1 Communications and Control Requirements

All commands to initiate SRV operation shall be provided from the Satellite Vehicle. After separation from the Payload, the sequence of spin, firing the deboost rocket, despin, and ejection of the thrust cone, shall be controlled by a timer installed in the SRV. After the re-entry phase, deployment of the parachutes shall be initiated by circuitry contained within the SRV. The SRV shall provide a 3 channel FM/FM telemetry system for supplying key event and environmental data. A Continuous Wave transmitter shall be provided as a beacon to assist in acquiring the capsule during recovery.

3.3.3.2.2 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. Environmental requirements are specified in IMSC T3-6-002.

3.3.3.2.3 Attitude Control Requirements

During the separation sequence, the Satellite Vehicle shall provide

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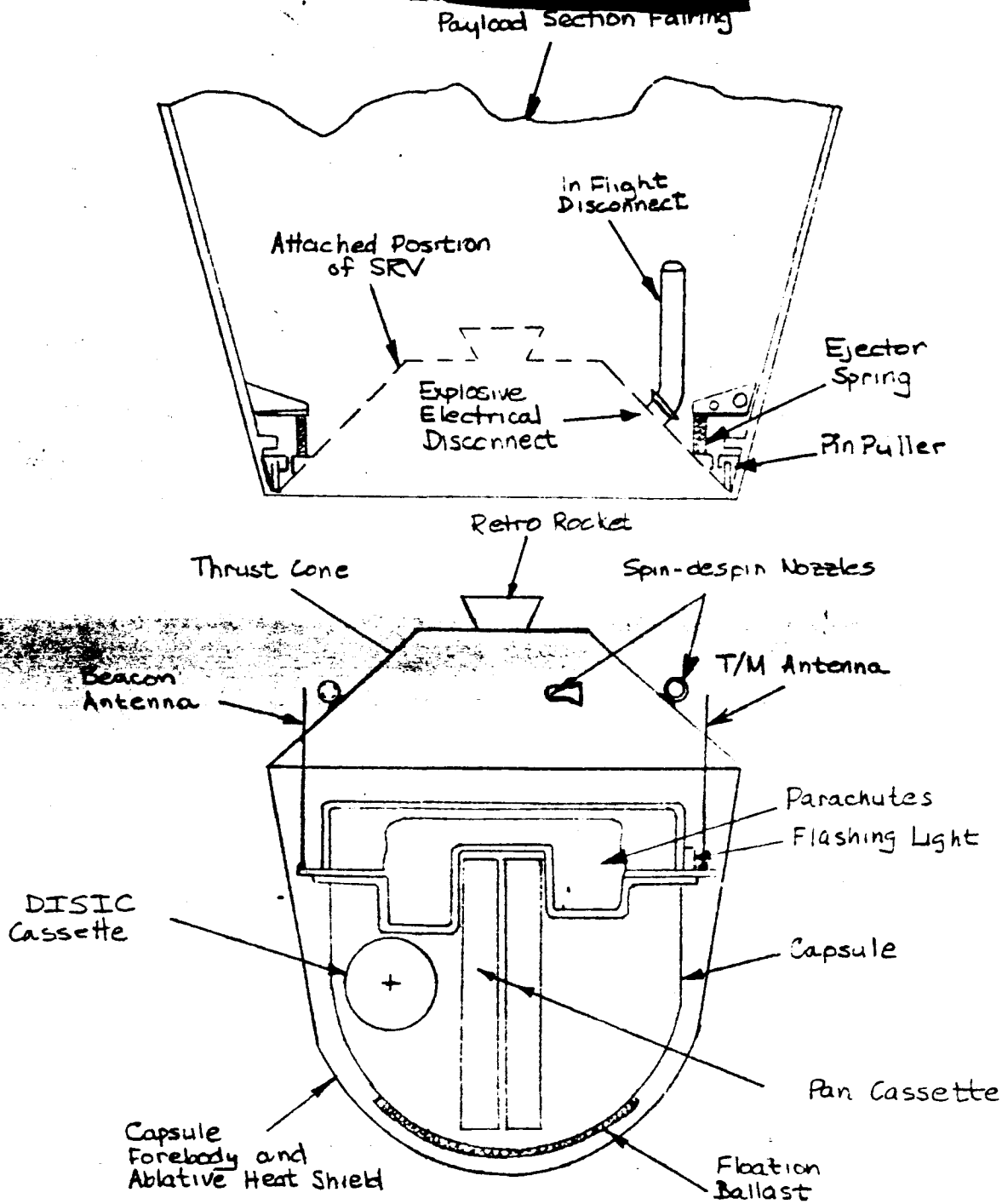


Figure 13

Satellite Re-entry Vehicle Inboard Profile

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The initial orientation of 120 degrees from the local horizontal. Subsequently, the SRV attitude shall be maintained during the retrorocket firing by spin stabilization. A spin rate shall be imparted to the SRV at a nominal time of 2.3 seconds after the separation command. After a time interval to accommodate retrorocket firing, the SRV shall be despun. 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.

3.3.3.2.4 Power Supply Requirements

During on-orbit operation, power to operate the film take-up cassette shall be provided by the Satellite Vehicle. The Satellite Vehicle shall provide power to accomplish activation of the SRV and separation from its payload 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. Electrical harnessing shall be provided either by the Payload Contractor or the SRV contractor to interconnect payload equipments located in the SRV.

3.3.3.2.5 Structural Design Requirements

Components in the capsule and on the thrust cone shall be packaged as necessary to provide space and access for payload components.

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 sinkport supplied with the capsule shall scuttle the capsule after 55 hours. Flotation shall not exceed 85 hours maximum. The capsule shall be capable of sustaining water impact while suspended on the parachute under conditions of a sea state of 3 with 18 knot surface winds. After water impact, the capsule shall float 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 230 lbs., excluding the parachutes. The desired rate of decent at

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10,000 ft. above mean sea level shall be less than 29.5 feet per second under standard atmospheric conditions. The main parachute canopy shall be designed for aerial recovery, with 120 to 230 lbs. suspended weight. Maximum aerial recovery altitude shall be 15,000 ft., and maximum aircraft speed shall be 135 knots indicated air speed.

3.3.3.2.6 Propulsion Requirements

Re-entry vehicle weight shall be a maximum of 400 lbs. The retro-rocket shall provide a total impulse of 10,500 lb. seconds \pm 3 percent. The retrorocket motor and all SRV pyrotechnic devices shall be compatible with the following requirements as related to Program [REDACTED]

A. Pyrotechnic Handling and Storage

Launch Site and Range Safety Regulations

Contractor Plant Safety Regulations

B. Service Life

Installation to activation time.

3.3.3.2.7 Instrumentation

The Satellite Recovery 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 coordinated with all affected contractors.

3.3.3.2.8 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 10 lumen 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 and batteries shall also be a minimum of 10 hours after water impact. In the event of failure of one (1) Recovery Battery, the remaining battery shall provide

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operation for a minimum of 5 hours. Operating life of the SRV telemetry subsystem and batteries shall be a minimum of 20 minutes after separation of the SRV from the Payload but with a 40 minute design goal.

Subsequent to recovery, simple and convenient means to turn off retrieval aids shall be provided.

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 SLV-2G THORAD booster shall be as follows:

Item	Weight (lbs)	Total Weight (lbs)
<u>Weight Empty</u>		7,795
Propellants	145,356	
Pressuriz. gas	794	
Solid Motor Boosters (3)	29,538	
<u>Stage I Weight at Liftoff</u>		183,483
Less Expendables	92,087	
Less Solid Motor Cases (3) (Burn out at 40 sec.)	4,809	
<u>Wt. at Solid Motor Separation (100 sec.)</u>		86,587
Less Expendables	77,095	
<u>Wt. at Main Engine Cutoff (218.4 sec.)</u>		9,492
Less Expendables	193	
<u>Wt. at Vernier Engine Cutoff (227.5 sec.)</u>		9,299

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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 .98 is to be utilized. The SLV-2G 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

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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 SLV-2G Thrust Augmented Thorad is a vertically launched, liquid-fueled space booster powered by a main gimballed rocket engine and three thrust augmenting solid propellant rocket motors. Pitch and yaw control is provided by gimbaling 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 for 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 attached 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 shall be 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.

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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 extent 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. 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. 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.

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

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- 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 a 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 liftoff until initiation of radio command guidance, the booster shall be controlled by an 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: ± 5 degrees in flight path angle, 9.6 nautical miles in position, and 450 ft. per second in velocity.

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

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<u>Event</u>	<u>Time from Liftoff (sec)</u>
Liftoff	0
Start roll to Launch Azimuth	2
Start pitch rate	9.4
Stop roll rate	(at programmed time)
Pitch rate step	35
Solid Motor Burnout	37
Pitch rate step	65
Pitch rate step	100
Drop Solid Motor Cases	102
Start "Dog leg" yaw rate (if req'd)	104
Stop yaw rate, start roll rate	120
Stop roll rate	124
Enable Radio Guidance Commands	135
Enable Main Engine Cutoff (MECO)	208
End Radio Guidance Steering	214
Main Engine Cut Off (Radio Command)	218
Vernier Engine Cut Off, Stop Pitch Rate	224
Separate from Satellite Vehicle (Radio Comm)	231

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 SLV-2G/SOLB Electrical Interface Specification.

3.3.4.3.2 . Propulsion Requirements

The propulsion subsystem of the SLV-2G 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.

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TABLE 10 Propulsion Performance Characteristics

Liquid Engine

Sea Level Thrust 172,120 pounds
Total Impulse - Vacuum 41,621,055 pound seconds

Solid Engines (3)

Vacuum Thrust 179,101 pounds
Total Impulse - Vacuum 6,476,306 pound seconds

All Engines

Total Impulse - Vacuum 48,097,361 pound seconds

Main Engine Burn Time 218 seconds

Vernier Engine Burn Time 227 seconds

Solid Motor Burn Time 40 seconds

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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)	172,120 pounds
Mixture ratio	2.15 \pm 2 percent
Specific Impulse (SL min)	248 sec
Propellant Utilization (min)	99.6 percent
Propellant Storage (min)	
Propellant Delivery (min)	

The booster shall contain valves, automatic sensors, relays, associated lines, and electrical circuitry to permit loading of fuel oxidizer and gas 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:

Axial Specific Impulse - Vacuum	271.8 seconds
Thrust (nominal during web-burn)	59,700 pounds
Total Impulse	2,158,176 pound 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 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

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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 SLV-2G/SOLB Electrical Interface Specification.

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

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 SOLB 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 SOLB, and enabling of the Agena velocity meter to control shut down of the Agena engine.

Controlled parameters during Agena guidance are orbital period, orbital inclination and flight path angle at injection. The steering commands shall cause 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.

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Normal preflight preparations require 8 days to generate necessary performance data, 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 the minimum possible 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.

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 Procuring Agencies.

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.

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I Interfaces internal to the [REDACTED] System, and accommodation provisions shall be as specified in the following subsections.

3.3.6.1.1 Payload Camera/Satellite 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.

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 DMEC Drawing 1324217, while the electrical interface is defined by T3-6-023.

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].

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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 SLV-2G and S01B vehicles is defined by LMSC Drawing 1376863. The electrical interface is defined on LMSC Drawing 1376862. The specification covering this interface is LMSC 1418956.

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 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 Test Operations Order, and the Recovery Group Operations Plan.

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3.3.7 Vulnerability Considerations

3.3.7.1 Modifications to Vehicle or Mission Plans

Any modifications to the vehicle or mission plans necessary to reduce vulnerability, in the event of hostile attack on the satellite, will be made only by approval of the [redacted] Program Directorate. However, the Program [redacted] contractors shall cooperate with the [redacted] [redacted] directorate in providing such information as necessary to aid them in developing survivability plans and retrofit hardware.

3.3.7.2 Command Link Considerations

The design and operation of the command link shall include consideration of the effects of deliberate attempts by a hostile agent to jam the links or gain control of the vehicle. Primary consideration should be given to gain of control which could result in unauthorized de-orbiting of the vehicle. Command Link vulnerability should be minimized consistent with other requirements.

3.3.7.3 Nuclear Hardening

As a design objective, new equipments electronic circuits should be designed to endure the level of radiation listed below without causing a catastrophic system malfunction. If this cannot be accomplished without penalty in system performance, a choice will be made by the [redacted] Program Directorate.

Levels

Prompt gamma radiation	10^8 rads/sec (10^{-7} second)
Integrated gamma dose	10^2 rads
Neutron Dose	10^{12} NVT

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

Normally, On-orbit control will be provided through the Tracking Stations under control of the U. S. Air Force Satellite Control Facility under the direction of the [REDACTED] Program Directorate for the vehicle and the PSAPO for the payload. In case of abnormal flight conditions or anomalies of the vehicle and/or payload, commands shall be subject to review, approval and control of the [REDACTED] Program Directorate. Operations for the [REDACTED] System shall be under the jurisdiction of the Flight Test Field Director (FTFD) FSCF 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 insofar 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 by [REDACTED] or PSAPO as appropriate. Support will be 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.

3.3.8.1 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.

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In addition, data processing equipment is used at IMBC (AP) for command generation, payload reporting and generation of on-orbit payload control data.

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 seventh parameter fit for average drag determination. Satellite spatial position errors on orbit for photographic command selection must be known, after sufficient target acquisitions, to:

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

C. Mission Programming

Mission programs will receive ephemeris, telemetry, payload capability, stored commands and alternative camera programs, and operational requirements including target search areas as inputs, and shall display the data in order to select commands necessary to control the camera operations.

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.

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3.3.8.2 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) shall be utilized for vehicle tracking. This 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 remaining 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 FPA 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 VTS 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 VTS 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

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with the TIM-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. The Shemya station shall provide telemetry recording and data processing.

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 TIM-18 and Telemetry and Data antenna. Antenna use will be determined by availability. The following antennas shall be used to support the [REDACTED] Program mission at the station locations noted:

<u>Station</u>	<u>Antenna</u>
[REDACTED]	TIM-18, Flat-Plane Array or Double-Helix
[REDACTED]	Double-Helix or Flat-Plane Array
[REDACTED]	Flat-Plane Array, Data Receiving or Double-Helix
[REDACTED]	Disc-on-Rod, or Double-Helix
[REDACTED]	TIM-18, Data Receiving, or Double-Helix
Shemya Auxiliary	Tri-Helix
[REDACTED]	Flat Plane Array or Double-Helix

The raw video output received by the ground station shall be routed to SubCarrier Discriminators (SCD's) 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. Decommunication capability of each of the SCF ground stations used for Program [REDACTED] is as follows:

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Station	Number of Decoms Available			
	30 pts/sec	60 pts/sec	20 pts/sec	10 pts/sec
[REDACTED] (on A & B systems)	(2)	(2)*	---	(2)
[REDACTED]	(2)	---	---	(2)
[REDACTED]	(2)	(2)*	---	(2)
[REDACTED]	(2)	---	---	(2)
[REDACTED]	(2)	---	---	(2)
Shemya Auxiliary	---	---	---	(1)
[REDACTED]	(2)	---	---	(2)

*A System Only

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:

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

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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. The payload shall comply to IMSC T3-6-002.

4.1 Design Reviews

Design reviews shall be performed for the purpose of evaluating the adequacy of design, design analysis, testing, and documentation. The Contractors shall submit a design review plan to their respective Procuring Agency for approval. Design reviews shall in general 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 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 appropriate procuring agencies shall be given adequate advance notice of all design reviews, and shall receive documented summaries of final system design reviews.

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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. A formal reliability program is not a part of the Payload effort.

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. The first form J-3 payloads will provide for adequate instrumentation to allow an engineering evaluation of the improved Payload design. The J-3 payload qualification system will also be maintained as an engineering test bed by the Payload Contractor for a period of nine months. With these exceptions however, engineering test and evaluation on the [REDACTED] System is expected to be minimal. 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 occurrence 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.

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

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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 equipment shall be qualified to the requirements of IMSC 6117B/D, except as specifically authorized. Payload equipment will be qualified to IMSC T3-6-002. Qualification test specifications, and all deviations from the above stated requirements, shall be reviewed and approved by the Procuring Agency. 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.

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.

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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 the equipment. 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 the [REDACTED] specifications 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 SOLB 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

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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 SLV-2G, Thrust Augmented Thorad, 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 SOLB. 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 acceptance 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 data, 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. Integration System Specifications, test plans and procedures shall be subject to review and approval by the Procuring Agency.

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

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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. Payload Subassembly Project Office (PSAPO)
- D. [REDACTED]
- E. Western Test Range
- F. Air Force Satellite Control Facility
- G. Ground Guidance Contractor - BTL/WECO
- H. Stage I Booster Contractor - DAC
- I. Satellite Vehicle Contractor - IMSC
- J. Payload Integrating Contractor - IMSC
- K. Supporting Organizations for the Launch Wing, Western Test Range, and Satellite Control Wing, such as Weather, Range Safety, etc.

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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.

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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, 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 Satellite Operations Center. 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.(LMSC A817971).

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 vehicle equipment are defined by LMSC 1414870, "Detail Specification S01B and SS01B Vehicles".

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6.1 List of Abbreviations

<u>Abbreviations</u>	<u>Full Terminology</u>
AC, ac	Alternating Current
AF	Air Force
AFSCF	Air Force Satellite Control Facility
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
DAO	Douglas Aircraft Company
dB, db	Decibels
DC, dc	Direct Current
deg	Degrees
DISIC	Dual Improved Stellar Index Camera
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
[REDACTED]	[REDACTED]
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
[REDACTED]	[REDACTED]

Abbreviations

Full Terminology (Cont'd.)

MLL	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
ORD	Orbital Requirements Document
OSFG	Orbital Sine Function Generator (Yaw Programmer)
PAM	Pulse Amplitude Modulation
PALC	Point Arguello Launch Complex
PDM	Pulse Duration Modulation
PPM	Pulse Position Modulation
PRD	Program Requirements Document
PSAPO	Payload Subassembly Project Office
PTU	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
TIM, T/M	Telemetry
TT & C	Telemetry, Tracking and Command
[REDACTED]	[REDACTED]

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Abbreviations

Full Terminology (Cont'd.)

UHF
USAF

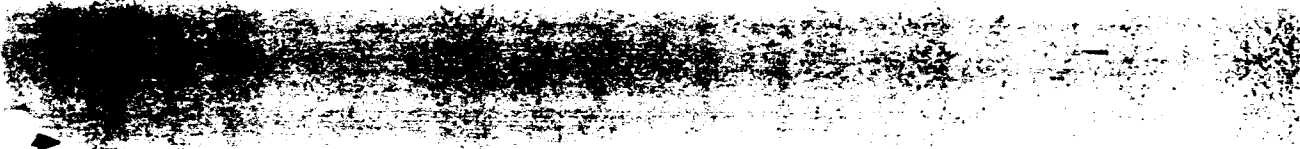
Ultra High Frequency
United States Air Force

VAFB
VCO
VHF
V/M


Vandenberg Air Force Base
Voltage Controlled Oscillator
Very High Frequency
Velocity Meter


WTR

Western Test Range



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ATTACHMENT #3

ERROR BUDGETS

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

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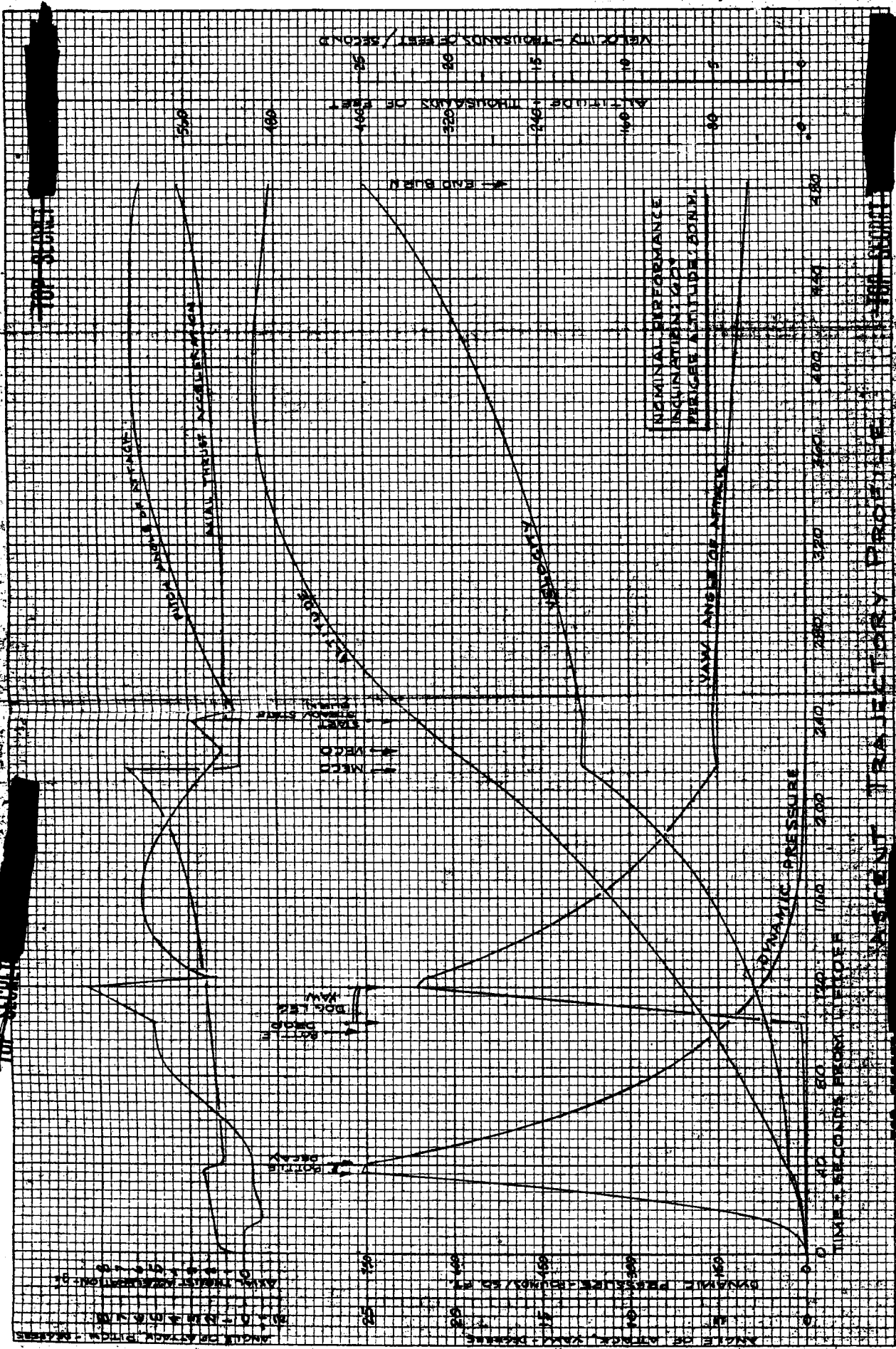


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