[redacted]

HANDLE VIA [redacted]
CONTROL SYSTEM ONLY

~~TOP SECRET~~/C

Spec. No. T3-0-012
Formerly: [redacted]
Page: 1 of 143 pages

Copy No. [redacted]

# [redacted] SYSTEM PERFORMANCE/DESIGN REQUIREMENTS

## MASTER SYSTEM SPECIFICATION

NOVEMBER 1969

APPROVED: [redacted]

APPROVED: [redacted]

APPROVED: [redacted]

Declassified and Released by the N R O

In Accordance with E. O. 12958

on NOV 26 1997

[redacted]

[redacted]

HANDLE VIA [redacted]

TABLE OF CONTENTS

TABLE OF CONTENTS (Contd)

## LIST OF ILLUSTRATIONS

LIST OF TABLES

## 1.0 SCOPE

This specification covers the requirements for the Corona J3[redacted]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 is also specified. This document is intended to specify the configuration of the Program as of November 1969.

## 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.

<u>Specifications and Standards</u>

| | |
|---|---|
| MIL-Q-9858A | Quality Control System Requirements |
| MIL-STD-785 | Reliability Program for Systems and Equipment Development and Production |
| MIL-STD-130C | Identification Marking of U.S. Military Property |

<u>McDonnell Douglas Astronautics, Western Division Documents</u>

| | |
|---|---|
| MDAC CP00155 | Production and Acceptance Specification SLV-2H |

<u>Lockheed Missiles and Space Company Documents</u>

| | |
|---|---|
| LMSC 1324217 | Satellite Vehicle/Payload Mechanical Interface |
| LMSC 1413451D | Performance Specification for AGE at Launch Complex (SLC-3W) |
| LMSC 1413878C | Performance Specification for AGE C12 |
| LMSC 1417293 | Interface Control Specification between SLV-2H/ Model 39205 Vehicle |
| LMSC 13[illegible]789 | SLV-2H/Agena Electrical Interface |
| LMSC 1336744 | SLV-2H/Agena Physical Interface |
| LMSC 1417675 | Detail Specification, Model 39205 Vehicle, Program |

[redacted]

<u>Lockheed Missiles and Space Company Documents</u> (Contd)

| | |
|---|---|
| LMSC T3-5-021 | DISIC Camera/AP Payload Interface Specification |
| LMSC T3-5-019 | Panoramic Camera/AP Interface Specification |
| LMSC T3-6-063 | Acceptance Test Specifications J-3 System |
| RO J3-001, RO J3-002 | Requirements Specification, J-3 System Payload |
| LMSC T3-6-002 | General Specification for Payload Qualification and Acceptance |
| LMSC T3-9-006 | Tracking, Telemetry, Commands, Power, and Pyrotechnic Interface Specification for A/P [redacted] Agena |
| LMSC T3-5-020 | SRV/AP Payload Interface Specification |
| LMSC T3-6-028 | Design Control Specification, J-3 System |
| LMSC A817971 | Special Programs Agena Secondary Payload Interface Specification |
| LMSC 447969B | Electromagnetic Interference Control Requirements and Electrical Interface for Agena Systems, Specification for |
| LMSC A376332 | Atmospheric Density Between 70 and 200 N.Mi. from Satellite Observations |
| LMSC 920493A | Interference Control Requirement Specification for Space Systems Ground Equipment (AGE) |
| LMSC B210541 | Program [redacted] Orbital Requirements Document, Mar 68 |
| LMSC 6117D or B | General Environmental Specification for Equipment of the Agena and Associated Payloads |
| LMSC A068386 | Vehicle Umbilical and Test Plug Assignments and Land Line Recording Requirements |
| LMSC A815752 | Thorad/Agena Range Safety Report for the Air Force Western Test Range |
| LMSC 220580, Rev. 1 | Safety Procedures Manual, LMSC Operations at Vandenberg Air Force Base, 14 March 1964 |

Lockheed Missiles and Space Company Documents (Contd)

| | |
|---|---|
| LMSC 224459 | Launch Stand Safety Plan |
| LMSC 226785 | Count Down Manual, VAFB SLC-1E |
| LMSC 226791 | Count Down Manual, VAFB SLC-3W |
| Not Numbered | Program [redacted] Explicit Guidance Equations |

Bell Telephone Laboratories/Western Electric Company Documents

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

Air Force Documents

| | |
|---|---|
| WTR No. 10900 | Program Requirement Document - Program [redacted] Headquarters WTR, Vandenberg AFB, Jan 67 |
| WTR No. 10900 | Program Support Plan, Program [redacted], Western Test Range, Mar 67 |
| AS-69-0000-02007 | Program [redacted] Orbital Support Plan, 4 Nov 68 |
| AFSCF 69-10 | Test Operations Order No. 69-10, Program [redacted] Det. 1 AFSCF, June 1969 |
| AFSCF-RCG No. 1-69 | Test Group Operations Plan, 6594th Test Group |
| Not Numbered | ARDC 1962 Standard Atmosphere |
| WTR S/N 00504 | Program Requirement Document |

Itek Documents

| | |
|---|---|
| DCS-397-1E | Design Control Specification, Pan Stereo Camera |
| ATS 397-1A | Acceptance Test Specification, Pan Stereo Camera |
| ATS 397-4 | Acceptance Test Specification, Film Supply Cassette |
| ATS 397-2<br>ATS 397-3 | Acceptance Test Specification, Take-Up Assembly |

Fairchild Documents

| | |
|---|---|
| SH64-23 | Design Control Specification, DISIC |
| 57DB-5B | Acceptance Test Specification, DISIC |

General Electric Documents

| | |
|---|---|
| GE S0010-02-0022 | System Test Specification Satellite Recovery Vehicle |
| GE S0040-01-0031 | Systems Acceptance Specification Satellite Recovery Vehicle |

## 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 the mission, the following events shall be performed:

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

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

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

D. Re-enter the recovery vehicles from orbit along a planned re-entry trajectory and impact at 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 payload data by recovery vehicles.

[redacted]

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.

DISIC system and DRCG information shall be provided to locate the vehicle orbital position at exposure within one quarter minute of arc at the local horizon in relation to geocentric earth coordinates, with a corresponding time duration within one millisecond.

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 (DISIC) 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 20 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 DISIC 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 Thrust Augmented Thorad Boost Vehicle (SRV-2HH and a Program [redacted] Model 39205 Agena to launch from the Western Test Range (WTR) into orbit. In addition to performing the function of second stage boost vehicle, the Agena shall serve as the Satellite Vehicle which supports and orients the photographic payload and re-entry vehicles. The re-entry vehicles containing the photographic record will be subject to air retrieval over water, or alternatively to water retrieval. Figure 1 illustrates the launch configuration for the [redacted] System.

The [redacted] System shall utilize flight qualified subsystems and/or components to the greatest extent feasible. On-orbit control will be performed utilizing the Satellite Control Facility.

3.3.1 Characteristics

3.1.1.1 Operational Characteristics

The [redacted] System encompasses the total capability necessary to achieve search-surveillance photography by an 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 provided 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 utilized to perform this function are the

~~TOP SECRET~~/C [redacted]



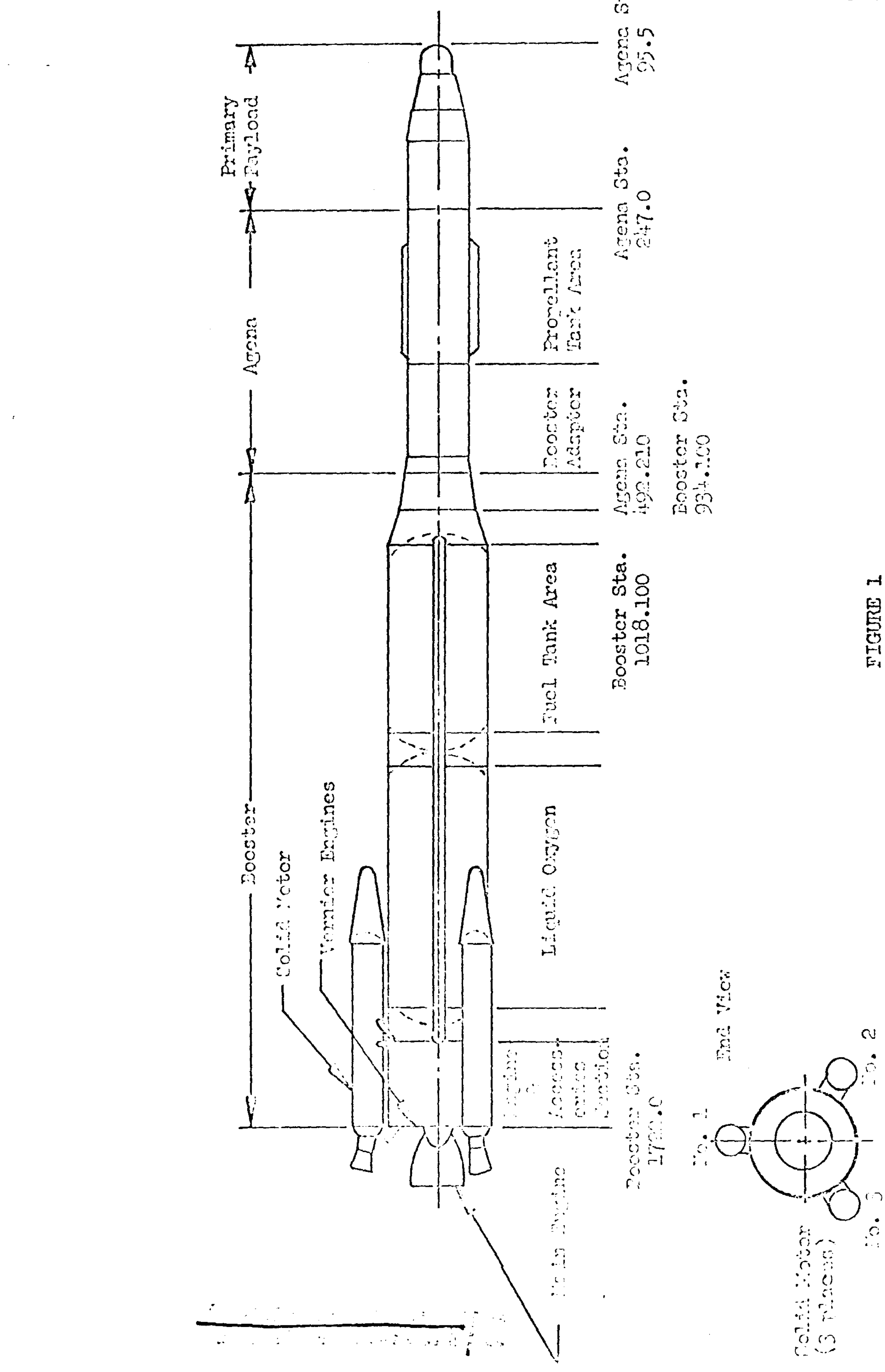



FIGURE 1

LAUNCH VEHICLE CONFIGURATION CORONA J3 [redacted] SYSTEM

~~TOP SECRET~~/C [redacted]

[redacted] Tracking Station [redacted], the [redacted] Tracking Station [redacted], the [redacted] Tracking Station [redacted], the Shemya Auxiliary Station (SRS), the [redacted] Tracking Station [redacted], the [redacted] Tracking Station [redacted], the [redacted] Tracking Station [redacted] and the [redacted] Tracking Station [redacted]. 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 shall be three.

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.

[redacted]

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

[redacted]

At appropriate times following completion of the orbital missions, the re-entry vehicles will be separated and ejected from orbit using their own deboost propulsion capability to impact in the selected retrieval area. The primary impact area is a broad ocean area within the WTR, located between 16° and 26°N latitude and 145° to 172°W longitude. The nominal impact latitude is 24°N for north to south recovery passes and 18°N for south to north recovery passes. Retrieval will be accomplished by air recovery as the primary mode, or by water recovery in the event that air retrieval is not accomplished. The STC shall compute impact predictions for use in commanding deboost of the Satellite Re-entry Vehicles, and for deployment planning by the recovery forces.

Under normal operating conditions, the vehicle and payload commands specified for the flight shall be implemented by the Flight Test Field Director (FTFD), Air Force Satellite Control Facility (AFSCF) on direction by the [redacted] Program Directorate for the vehicle and the Photographic Reconnaissance Systems/West Coast Project Office (PRS/WCPO) for the payload. In the case of abnormal flight conditions or anomalies of the vehicle and/or payload, commands shall be subject to review, approval, and control of the Program [redacted] 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 commands will be based on the payload data available generated by the PRS/WCPO. 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 [redacted] launches.

B. Launch Operations

Program [redacted] Vehicles shall be launched from Vandenberg Air Force Base. Launch operations will be under the cognizance of the 6595th Space Test Group. Launch complex SLC-3W will be utilized to mate, checkout and launch the SLV-2H/Agena Boost vehicle. SLC-1E complex will be used as a backup facility with 60 day notice required for activation.

Program [redacted] operations will be supported by the SAMTEC 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 Air Force Base (VAFB) will be utilized as available and necessary for the implementation of the program.

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

C. Recovery Operations

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

The recovery sequence is divided into two phases: (1) re-entry and (2) 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. Deboost shall be achieved by means of spin system, a retro-rocket, and de-spin system. Immediately after de-spin, the thrust cone (mounting platform for the rocket and spin system) 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 60,000 to 74,000 feet.

The recovery force will consist of aerial recovery aircraft equipped with electronic detection and direction finding equipment. A minimum of four JC-130 type aircraft shall normally be deployed in a North to South direction in accordance with the impact prediction provided by STC. In addition, one flyable spare aircraft must be available either airborne or on the ground. The aircraft shall be equipped with special air retrieval gear to snare and secure the capsule/chute during its descent. The recovery force will also employ surface vessels with tracking/direction finding equipment and helicopters to retrieve a capsule that impacts the sea. Additional support 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 nineteen (19) days after launch. 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 nine to ten 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 SCF 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 6594th Test Group (TG) Honolulu, Hawaii. Logistic support will be rendered by Pacific Air Forces Base Command (PACAFBASECOM). 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 office of [redacted] Until such time as the designated courier is able to assume physical custody, the Command 6594th TG will be responsible for the capsule's physical and security safeguarding per designation by the Commanding AFSCF. Upon assuming physical custody, the courier shall be responsible for capsule handling and security. Transportation and delivery to the designated processing center is the responsibility of the [redacted].

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 search-area considerations, ground track synchronization, and performance available from the launch vehicle system. Parameters of primary importance are the orbital period, perigee altitude, argument of perigee and orbital inclination usually in the aforementioned order. With the SLV-2H/Agena 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 115 degrees
   (Most probable inclinations: 75 to 85 degrees)

2. Range of perigee altitude: 80 to 200 n.m.

(Most probable perigee altitude: 85 to 100 n.m.

3. Range of orbital period: 88 to 91.5 minutes

4. Range of perigee location: $90^{\circ}$N to $90^{\circ}$S latitude

(Most probable perigee location: $20^{\circ}$N to $60^{\circ}$N latitude)

Vehicle structural and performance limitations may preclude flying all possible combinations of the above parameters.

Accuracy Limits. Within the range of missions specified above, the following 3 sigma tolerances shall not be exceeded:

| | |
|---|---|
| 1. Inclinations: | Plus 0.30 deg. |
| | Minus 0.30 deg. |
| 2. Perigee Altitude: | Plus 15 N.Mi. |
| | Minus 15 N.Mi. |
| 3. Period: | Plus 0.45 min. |
| | Minus 0.45 min. |
| 4. Argument of Perigee: | |
| (a) For eccentricities of 0.008 or less: | Plus 180 deg. |
| | Minus 180 deg. |
| (b) For eccentricities greater than 0.008: | Plus 80 deg. |
| | Minus 30 deg. |

B. Ascent Requirements

Program [redacted] launches using the SLV-2H/Agena vehicles will be conducted from the following launch sites of Vandenberg Air Force Base:

| Complex | Pad Azimuth from North | Geodetic Latitude | Longitude | Elevation above SL |
|---|---|---|---|---|
| SLC-3W | 223.53° | 34.643653° | 120.59303° | 435 ft. |
| SLC-1E | 218.50° | 34.756153° | 120.62522° | 160 ft. |

SAMTEC 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 80 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 80 degrees the dog-leg maneuver must be accomplished after the predicted down-range impact has passed the critical range safety boundary.

The Launch Window shall normally be one hour with the launch generally scheduled with opening of the window. The optimum launch time shall be computed for each flight on the basis of required ground search-area lighting conditions, daylight in the recovery area, 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, satellite vehicle temperature-sensitive equipment, horizon sensor ascent look angle, and solar array alpha 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 91.5 minutes.

| | Time (Sec) | Down-Range Distance (N.M.) |
|---|---|---|
| Launch | 0 | 0. |
| Solid Motor Burnout | 40 | .51 |
| Solid Motor Separation | 102 | 9.8 |
| Booster Main Engine Cutoff | 218 | 123.7 |
| Vernier Engine Cutoff | 227 | 142.5 |
| Booster Separation | 231 | 168.0 |
| Optical Door Ejection | 233.5 | 168.7 |
| Agena Engine Ignition | 240.5 | 170.5 |
| Solid Motor Impact | 369.76 | 21.78 |
| Booster Impact | 642.8 | 813.85 |
| Agena Engine Cutoff | 483 | 869.0 (Orbit injection |

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 Satellite Operations center. For the SLV-2H/Agena vehicles the following typical 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 Lifeboat 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.
11. Solar array alpha angle adjustment.
12. Agena pre-programmer ascent pitch rates.

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 25 calendar days.

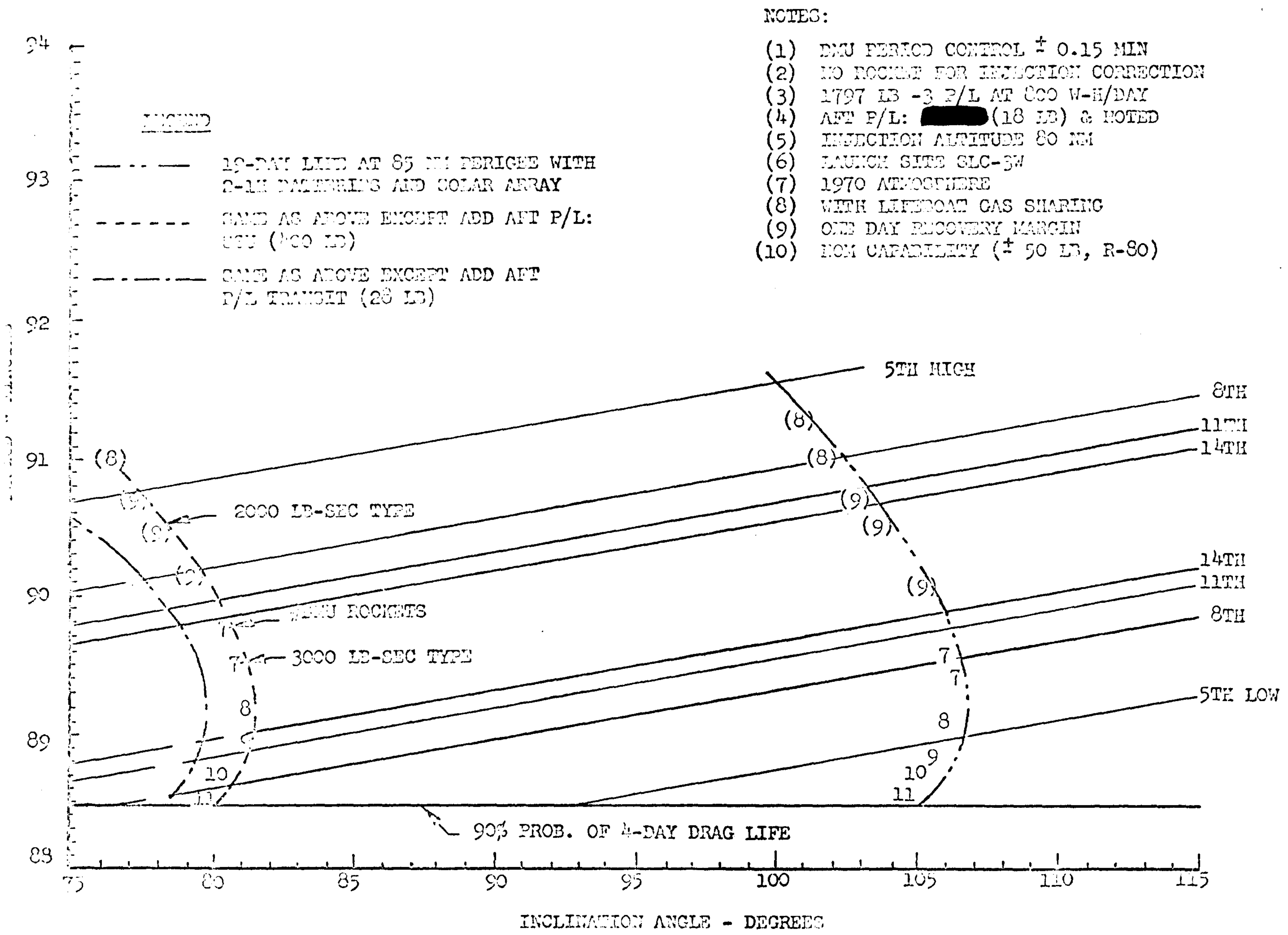
C. <u>Mission Performance</u>

Nominal <u>Mission Duration</u> shall be 19 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. Figures 2 and 3 illustrate the nominal mission capability of the system using SLV-2H boosted vehicles launched from SLC-3W. Perigee altitude shall be nominally 85 to 100 nautical miles. The mission parameters are given in terms of orbit plane inclination, orbital period, and constant lines representing the number of days at which the

FIGURE 2

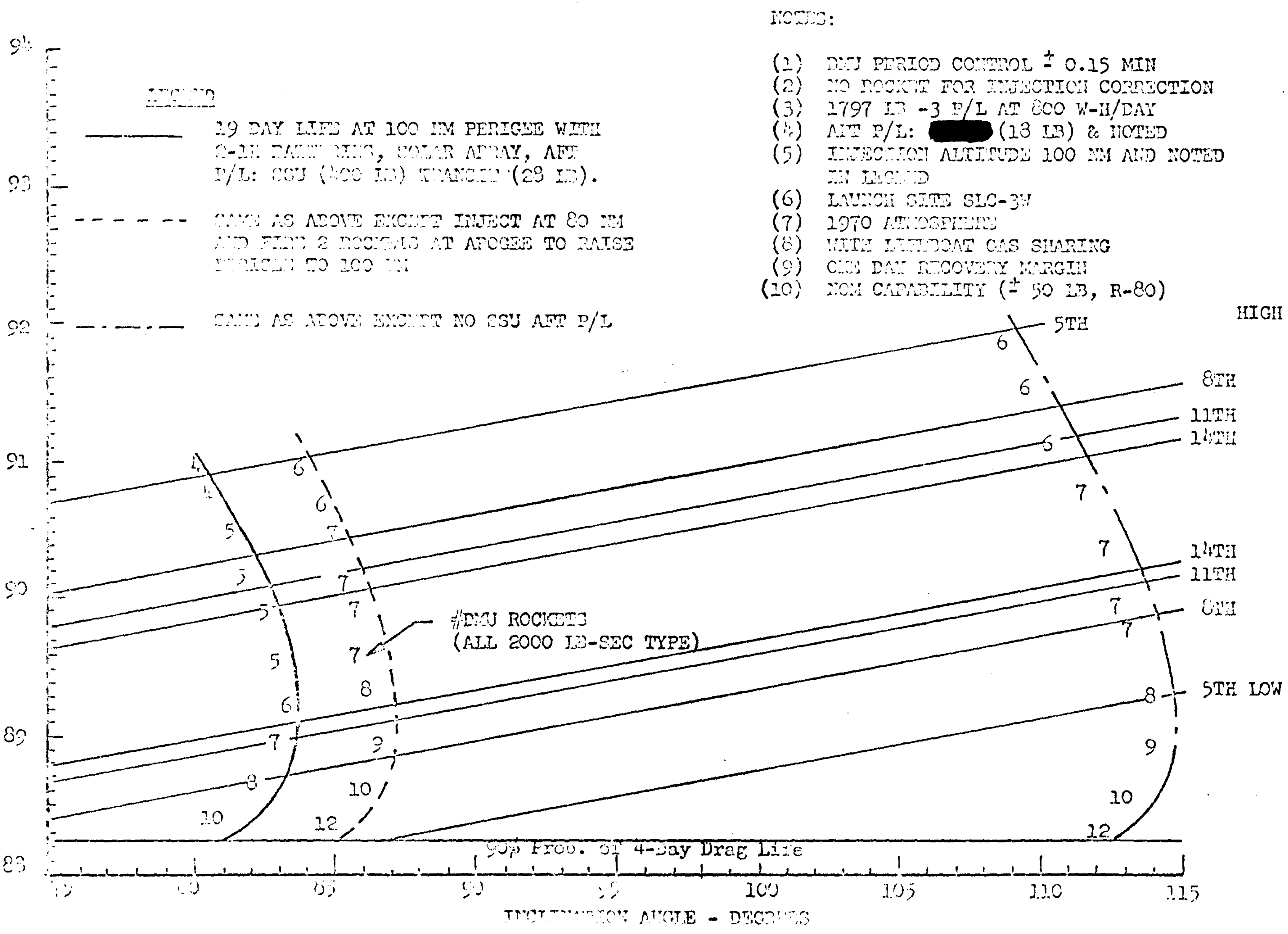
ESTIMATED THORAD/AGENA CAPABILITY FOR LOW ALTITUDE MISSION PLANNING



[redacted]

FIGURE 3

ESTABLISHED THORAD/AGENA CAPABILITY FOR HIGH ALTITUDE MISSION PLANNING




[redacted]

satellite point will repeat the ground trace initially covered. Repeat of the ground trace is termed orbit synchronization. The missions that nominally can be flown are shown on Figures 2 and 3. The missions that can be flown are to the left of the appropriate mission line.

Orbit Sustenance 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. The operating regime with near-circular (low eccentricity) orbits of 85 to 100 nautical miles offers significant payload advantages of improved scale, more constant compensation of image motion, and increased opportunities for payload operation on Northbound as well as Southbound passes. The desired synchronization can be attained by flying the shorter period orbits (Eastward closure). However, a satellite vehicle orbiting at these lower altitudes will be noticeably affected by the atmospheric environment and may require an orbit sustenance capability to make-up the velocity decrement caused by drag.

A drag makeup (DMU) 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 which are available in two sizes shall have a nominal impulse of either 2000 or 3000 pound seconds per rocket.

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

The primary latitude for the Hawaii recovery zone shall be 24 degrees North on North to South passes 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.
7. Uncertaintities in actual ballistic parameter, atmospheric density and surface winds.
8. Electrical or payload material not separating properly inducing a torque to the SRV.

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

| | |
|---|---|
| Orbit Inclination | 80 degrees |
| Perigee Height | 100 N.M. |
| Perigee Lat. | $45^{o}$N |
| Period | 90 minutes |

| Guidance & Control Subsystem | Direction | Dispersions (N.M.) Up-range | Down-range | Cross-range |
|---|---|---|---|---|
| Primary | N to S | 60 | 75 | ± 12.5 |
| Primary | S to N | 140 | 190 | ± 15.5 |
| Backup | N to S | 220 | 540 | ± 40 |

Abort of the launch or the orbital phases of the missions shall not cause the nature of mission objectives to be revealed to unauthorized persons. In the event of an aborted launch, provisions shall be made to recover payload equipment under appropriate security conditions. 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 occur in orbit, any subsequent SRV re-entry due to drag will cause the SRV (with payload) to break up.

Existing Range Safety Requirements specified in LMSC A815752 shall be complied with.

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

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

| | | |
|---|---|---|
| 1. | Scheduled countdown to launch | 0.95 |
| 2. | Launch and booster ascent | 0.97 |
| 3. | Agena separation and orbit injection | 0.98 |
| 4. | Orbital mission (1 day active) | 0.98 |
| | (10 days active) | 0.93 |
| | (20 days active) | 0.86 |
| 5. | Re-entry and recovery (each SRV) | 1.00 |

On-Orbit Disposal of the satellite vehicle and payload equipment shall occur by the natural re-entering of the vehicle into the earth's atmosphere due to drag effects, with subsequent destruction of the vehicle by re-entry heating and aerodynamic loads.

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 Special Programs Director (SPD) is responsible for 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 designated processing facilities.

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

The Director, PRS/WCPO, is responsible for total payload sub-assembly development, production (excludes procurement of the DISIC) and test; the provision of the software support to the Satellite Operations Center before, during, and after missions; operation of the LMSC-AP facility; and adherence to master system specifications; and interface specifications.

By definition, the Corona Payload Sub-assembly includes the Corona panoramic cameras and DISIC, film transport mechanisms, the SRV's, supporting structure and shell, and those other items normally installed and tested at the LMSC-AP facility.

In addition, the PRS/WCPO 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 at the appropriate time

[redacted]

that the Payload Sub-Assembly is ready and to act as the Payload Sub-Assembly Assistant to the SPD.

<u>Launch Vehicle Contractor</u> - The Launch Vehicle Contractor shall provide the Stage I Booster vehicles and will also provide all services necessary to checkout and launch the vehicle. McDonnell Douglas Astronautics Company, Western Division (MDAC), is the Launch Vehicle Contractor for the SLV-2H Thrust Augmented Thorad used for Program [redacted]

<u>Ascent Guidance Contractors</u> - The Guidance Contractors shall provide the guidance equations, guidance-computor programming, and airborne guidance equipment necessary to inject the Satellite Vehicle into specified orbits within allowable tolerances. Radio commend guidance will be utilized to steer both the Stage I booster and the Satellite Vehicle during ascent. Western Electric Company/Bell Telephone Laboratory and [redacted] are the Guidance Contractors for Program [redacted]

<u>Satellite Vehicle Contractor</u> - 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 Agena Model 39205 for Program [redacted]

<u>Payload Contractor</u> - The Payload Contractor shall provide the vehicle payload section, and shall integrate the installation of camera equipment, Satellite Recovery 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 PRS/WCPO. 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, however, the Contractor shall provide field engineering support, Fairchild Camera and Instrument Corporation (FCIC) is the DISIC Contractor.

Film Contractor - The film contractor shall be responsible for manufacture of the film, necessary checkout, and loading of the supply spools [redacted] 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 Space Test Group 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 by the U. S. Air Force Satellite Control Facility operating from the Satellite Test Center at Sunnyvale, California. The AFSCF shall be responsible within the direction of the [redacted] Program Directorate and the PRS/WCPO 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 6594th Test Group as designated by the Commander, AFSCF.

Integrating Contractor - The Integrating Contractor shall be responsible to the [redacted] Program Directorate for the 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. 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]

[redacted]

TOP SECRET/C



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 issuance of spares. Such controls shall provide for considerations of controlled life items and for-re-acceptance testing of spare equipment where extended storage periods may be incurred.

3.1.1.3 Personnel and Training

The number of personnel, personnel prerequisities, and required training needed to support the [redacted] System shall be identified by each major contractor and participating Air Force Organization. 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.

[redacted]

### 3.1.2 System Definition

#### 3.1.2.1 System Engineering Documentation

##### 3.1.2.1.1 System Functional Flow Diagram

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

#### 3.1.2.2 Functional Subsystem List

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

##### 3.1.2.2.1 Satellite Vehicle Subsystem

The Satellite Vehicle Subsystem shall consist of an Agena Model 39205 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-2H first stage booster vehicle with the Agena Model 39205 serving as the second stage booster. Ascent guidance shall be provided by on-board autopilots in conjunction with 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.

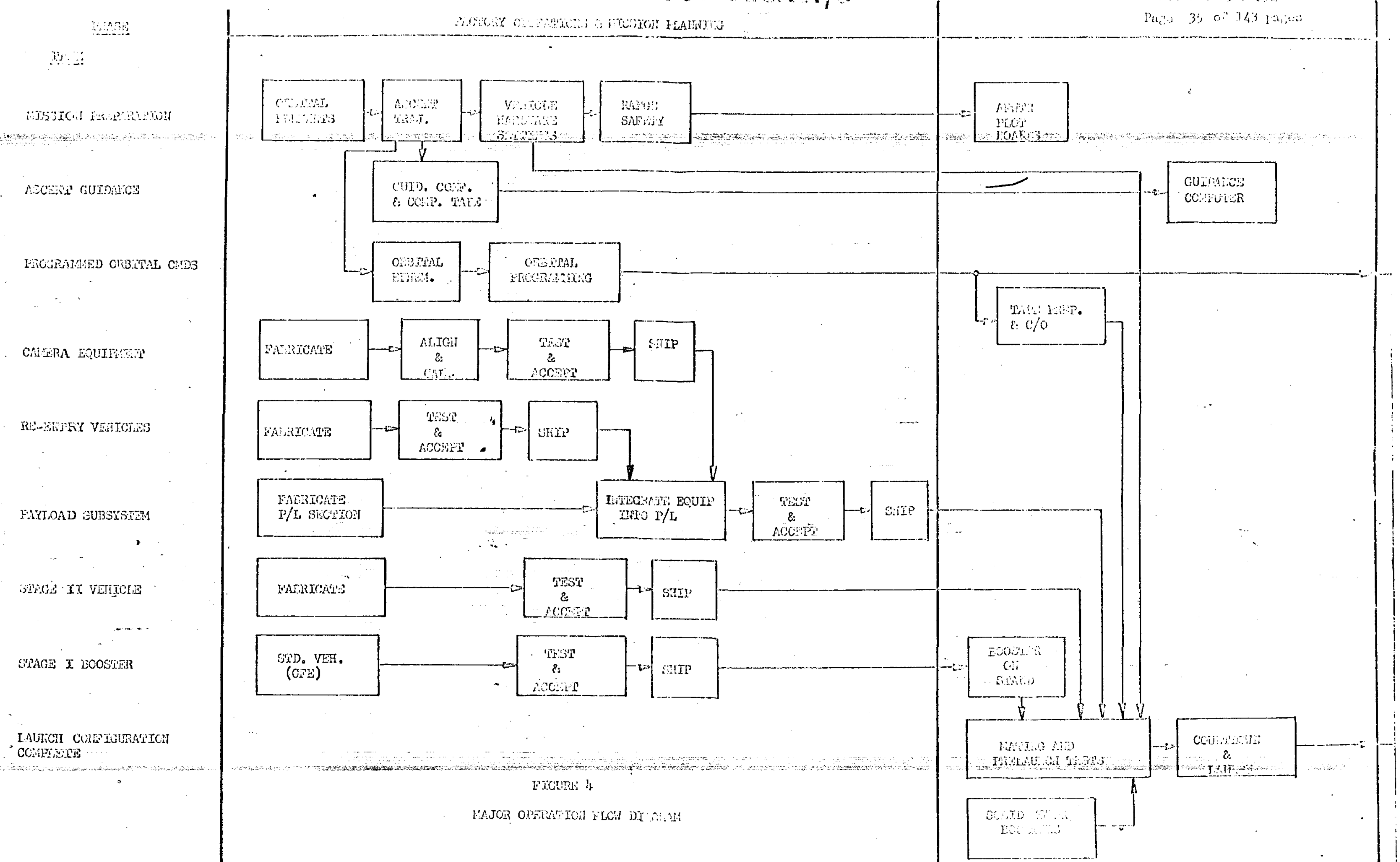



FIGURE 4

MAJOR OPERATION FLOW DIAGRAM

- CORONA J3 [redacted] SYSTEM PERFORMANCE AND DESIGN REQUIREMENTS MASTER SYSTEMS SPECIFICATION
  - SYSTEM TEST PLAN [redacted] PROGRAM
    - PAYLOAD TEST PLAN J3 SYSTEM
    - VEHICLE TEST PLAN AGENA MODEL 39205
    - VEHICLE TEST PLAN THORAD SLV-2H
    - PERFORMANCE SPECIFICATION FOR AGE AT SLC-3W COMPLEX LMSC [illegible]
    - PERFORMANCE SPECIFICATION FOR AGE C-12
    - PROGRAM REQ. DOC. NATIONAL RANGE DIV. WTR S/N 00504
    - PROGRAM SUPPORT PLAN PROGRAM [redacted] WTR 10900
    - THORAD/AGENA RANGE SAFETY REPORT FOR AFWTR LMSC A615752
    - COUNTDOWN MANUAL PROGRAM [redacted] LMSC 226791
    - ORBITAL REQ. DOC. PROGRAM [redacted] LMSC B210541
    - ORBITAL SUPPORT PLAN PROGRAM [redacted] No. 10900
    - TEST OPNS. ORDER PROGRAM [redacted] AFSCF 69-10
    - TEST GROUP OPERATIONS PLAN 1-69

FIGURE 5
OPERATION REQUIREMENTS & PLANS

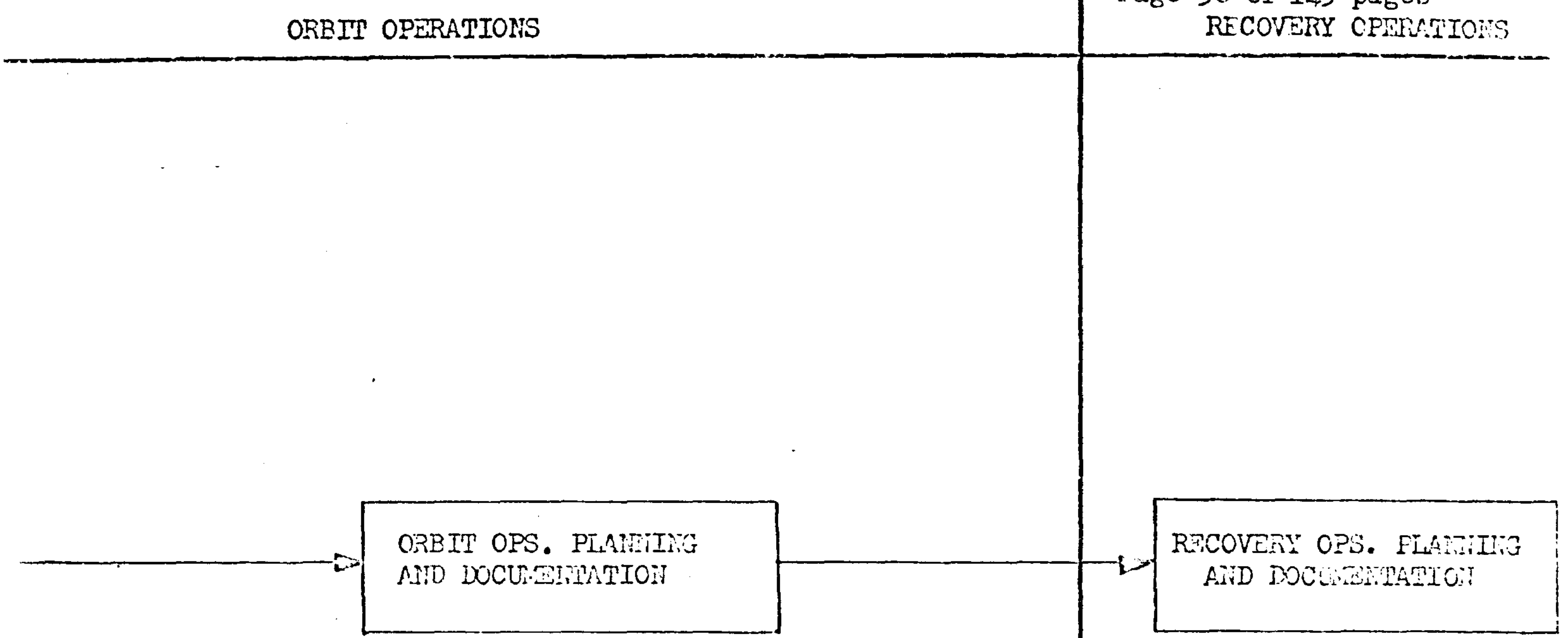
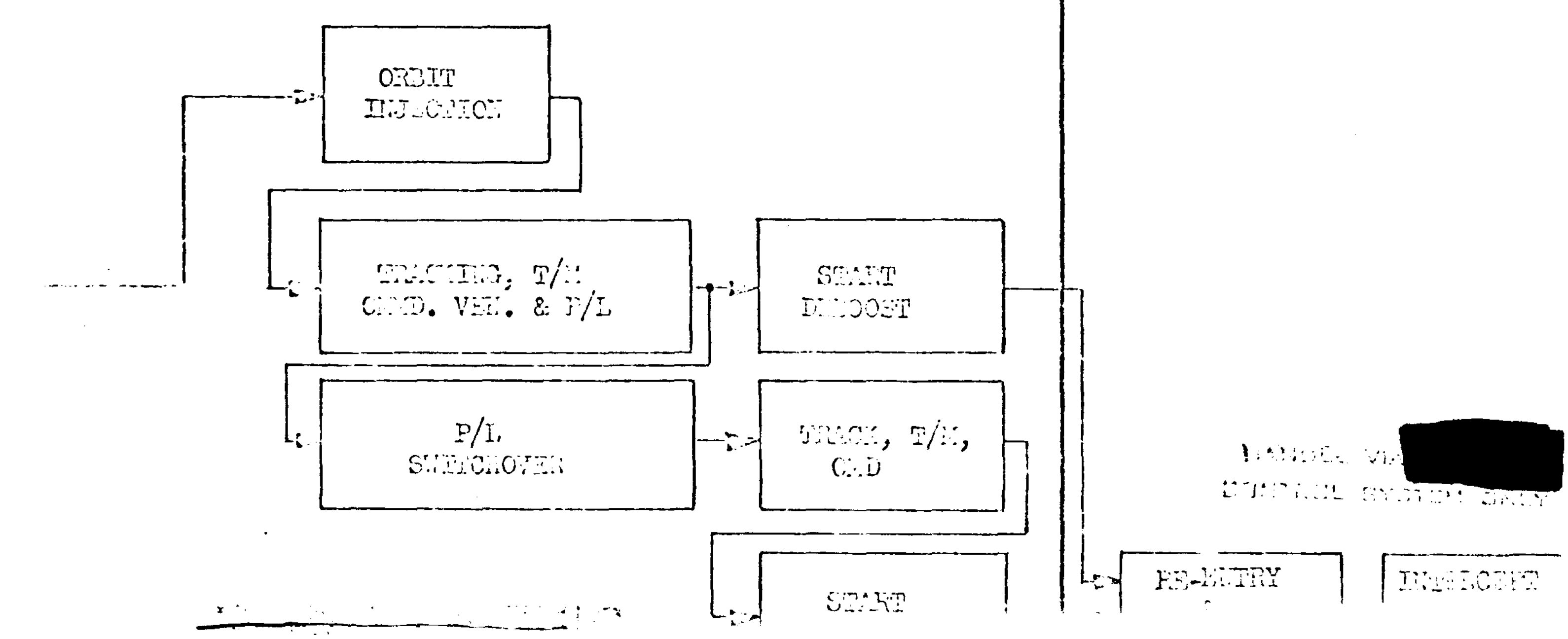



Figure 4 (Continued from preceding page)




HANDLE VIA [redacted]
CONTROL SYSTEM ONLY

3.1.2.2.3 Mission Control and Communications 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.

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 exceeding the specified 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.

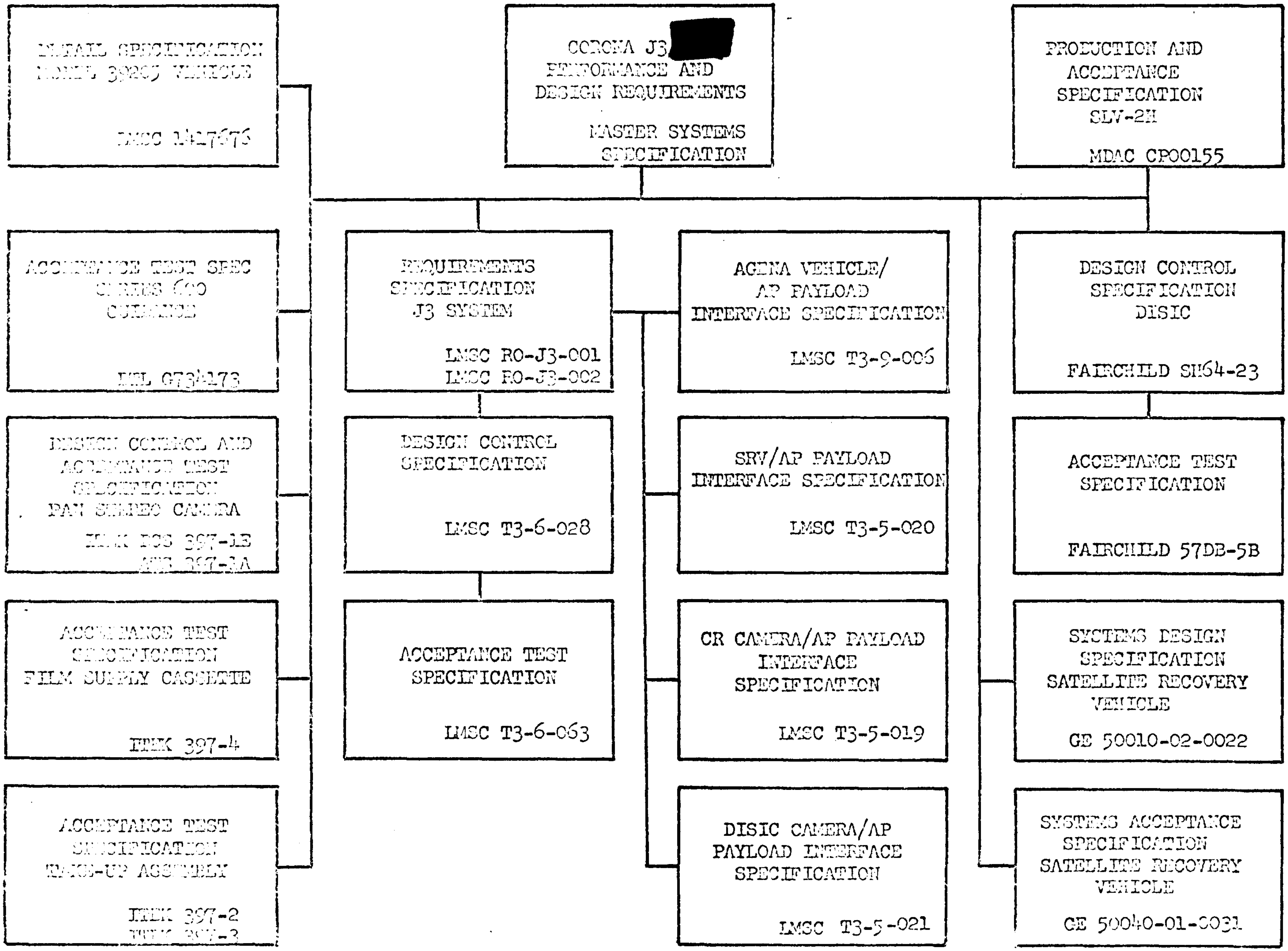



FIGURE 6. SPECIFICATION TREE - MAJOR END ITEMS

[redacted]

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-wearout cyclical life be less than three times the maximum estimated or experienced Program [redacted] yclical 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 categories: (1) Reliability Design Objectives and (2) 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 Agena, Model 39205 (orbital mode, 20 days maximum) active operation — Design Objective 0.86

B. Payload Equipment (20 day operation) — Design Objective 0.93

C. Re-entry Vehicles — Design Objective 1.00

3.1.3.1.2.2 Launch Vehicle Subsystem

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

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, Model 39205, Ascent Mode)

Design Objective 0.95

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 1.00 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 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 turn-around time of 42 working days from launch to launch or 25 days for emergency turn around time.

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

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

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

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

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 [redacted] 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 liftoff. Testing for pre-launch environment compliance shall be limited to humidity testing. The following climatic environments are associated with the pre-launch operations:

A. Temperature - Surrounding air temperature from a minimum of $25^{\circ}F$ to a maximum of $100^{\circ}F$.

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 seashore conditions.

E. Sunshine - Air conditioning shall be used to control the Agena and primary payload temperature on the pad.

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.

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, protective coating or replacement shall be employed.

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

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

Maximum Ground Wind Condition - Including Gusts

| Condition | Vehicle Stage | Propellant Tanks | | Max. Velocity in Knots | | |
|---|---|---|---|---|---|---|
| | | Pressurized | Loaded | Unbolted (Toppling Condition) | Bolted at Base | Bolted at Base and Supported at STA. 978 (SLC-1E) STA. 1151 (SLC-3W) |
| A | Booster | No | No | 14 (1) | 44 (1) | 75 |
| | Orbital | Yes | No | 25 | 44 | |
| B | Booster | No | No | 29 | 42 | 75 |
| | Orbital | Yes | Yes | | | |
| C | Booster | No | Yes | 37 | 37 | 75 |
| | Orbital | Yes | Yes | | | |
| D | Booster | Yes | Yes | 43 | 43 | 75 |
| | Orbital | Yes | Yes | | | |

(1) Solid fuel booster rocket not attached.

NOTE: In the event of an abort during countdown, after detanking the booster (Thor) must be pressurized to 15 PSI if wind exceeds 44 knots.

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. Agena Environment - The Agena shall undergo an environment as described in LMSC 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 LMSC 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 (760 mm of Hg) to a pressure of $1 \times 10^{-8}$ 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 structure, control, or range safety capability to be exceeded. Prior to launch, the Integrating Contractor shall compute the wind shear forces utilizing the winds aloft data from standard Rawinsonde observations, or special fast rising balloons, and provide the Space Test Group with percentages of structural control capability and ascent ground tracks for drift. Standard observations will be accomplished at T-12, T-6, and T-1 1/2 hours from liftoff. If sufficient safety margins cannot be attained because of the winds aloft, the launch shall be delayed.

E. Atmospheric Density Model - The ARDC 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 levels that are expected to exist on a SLV-2H/Agena vehicle are tabulated below. These sound pressure levels are expected to last no longer than ten seconds, and are measured at the base of the booster adapter.

| Frequency Range | Exterior SPL (db)** | Interior SPL (db)** |
|---|---|---|
| 18.8 - 9600 | 158 | 145 |
| 18.8 - 37.5 | 140 | 125 |
| 37.5 - 75 | 143 | 129 |
| 75 - 150 | 146 | 132 |
| 150 - 300 | 149 | 137 |
| 300 - 600 | 152 | 137 |

| Frequency Range | Exterior SPL (db)** | Interior SPL (db)** |
|---|---|---|
| 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 - The axial acceleration oscillatory component is expected to occur approximately 10 seconds before MECO and be of a 0 to 3.9 peak acceleration (3 sigma) with a frequency range of 11 to 21 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:

| | | |
|---|---|---|
| A. | Vacuum: | $1.5 \times 10^{-7}$ to $10^{-10}$ mm of Mercury |
| B. | Solar Radiation: | 445 BTU/ft$^2$ hr (nominal) |
| C. | Earth Shine: | 68.7 BTU/ft$^2$ hr (nominal) |
| D. | Earth Albedo: | 38% of the solar energy (nominal) |
| E. | Magnetic Field: | 560 milligauss at the poles to 260 milligauss at equator for an altitude of 125 n.mi. |
| F. | Atmospheric Density: | The ARDC 1962 Std. Atmosphere, modified per LMSC A376332 |

For 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 1952 shall be considered as representative of atmospheric conditions for the descent environment of analysis purposes. Winds aloft, cloud cover and visibility conditions shall be monitored in the recovery area during Program [redacted] operations. This meterological 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 degrees 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.

*(Sea state as defined in U.S. Navy Hydrographic Office Buford Scale with corresponding Sea State code chart.)

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).

3.1.3.4 Transportability

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

3.1.3.5 Safety

Normal transport, handling, and pad safety requirements will be recognized in the design and handling of Program [redacted] vehicles, equipments, and components. Compliance shall be required with applicable regulations for pressurized lines and bottles, explosives, and toxic propellants.

(Reference LMSC 220580 Safety Procedure Manual and LMSC 22445B Launch Stand Safety Plan.) Particular attention shall be given to selecting explosive ordnance which will meet the Western Test Range criteria (AFWTRM 127-1) for radio-frequency-electromagnetic environments.

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

3.2 System Design Control and Contruction Standards

3.2.1 General Design and Construction Requirements

3.2.1.1 Specifications and Standards - The selection of specifications and standards pertinent to the design and construction of vehicles, subsystems, equipment and components of the [redacted] System shall be identified in the appropriate end item specification prepared by the Contractor. Specifications and standards selected, and the degree of imposition on the design and construction of System [redacted] end item hardware, shall be subject to review and approval by the [redacted] Program Directorate, except as provided in paragraph 3.1.1.1.2 A.

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. Requirements for materials, parts, and processes shall be stated in the Contractor's specifications.

3.2.1.3 Moisture and Fungus Resistance - Moisture and fungus resistance shall be compatible with the environments outlined in Sections 3.1.3.3.1 and 3.1.3.3.4. Requirements for moisture and fungus resistance shall be stated in the Contractor's specification.

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.

3.2.1.7 Electromagnetic Interference

All Agena systems shall conform to the requirements of paragraph 3.2 of LMSC 447969B, "Electromagnetic Interference Control Requirements and Electrical Interface for Agena Systems, Specification for". New equipment shall be designed using LMSC Design Handbook techniques for controlling EMI including grounding, bonding, shielding, cable routing, and filter application requirements. Testing need not be conducted on equipment electrically similar to equipment already successfully flown on previous Agena missions. In cases where the equipment tested does not meet the requirements of Para. 4.3 of LMSC 447969B, the Responsible Engineer shall prepare a report describing the modifications which would be necessary to achieve full compliance. Included in the report shall be a statement of whether the out-of-spec conditions will cause adverse effects or malfunctions or any other part of the vehicle equipment and system.

The primary payload shall conform to LMSC T3-6-002, "General Specification for Payload Qualification and Acceptance".

New AGE equipment shall be designed using LMSC 920493A, "Interference Control Requirement Specification for Space Systems Aerospace Ground Equipment (AGE)" as a guide.

Whenever new EMI susceptable or EMI generating systems or components are incorporated into the vehicle system, a compatibility test will be run at the direction of [redacted] CSE to determine overall system functional integrity.

3.2.1.8 Identification and Marking

The identification marking requirements for vehicle equipment and components shall be in accordance with MIL-STD-130C. 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 Agena Model 39205 for Program [redacted] issions shall be as follows for the conditions stated. These weights do not include the mission payload or SRV weights.

## NOMINAL 846 AGENA WEIGHT SUMMARY

| | Weight (lbs) | Total Weight (lbs) |
|---|---|---|
| Weight Empty | | 2176 |
| Propellants | 13520 | |
| Helium | 1 | |
| Attitude Control Gas (-5 Mix) | 154 | |
| Auxiliary Attitude Control Gas - L/B (-5 Mix) | 21 | |
| 2 1H Batteries | 249 | |
| 12 DMU Rockets | 192 | |
| Gross Weight Without Payloads | | 16313 |
| Less Adapter and Attach | 394 | |
| Less Retro Rockets | 10 | |
| Less Destruct System | 6 | |
| Less Horizon Sensor Fairings | 7 | |
| Less Attitude Control Gas | 1 | |
| Ignition Weight Without Payloads | | 15895 |
| Less Propellants | 13362 | |
| Less Engine Start Charge | 1 | |
| Less Attitude Control Gas | 3 | |
| Burnout Weight | | 2529 |
| Less Residual Propellants | 56 | |
| Less Helium | 1 | |
| Propellant Contingency | 102 | |
| Weight on Orbit with Gas but Without Payload | | 2370 |
| Less Remaining Attitude Control Gas | 150 | |
| Less Remaining L/B Gas | 21 | |
| Empty Weight on Orbit Without Gas, Without Payload, with 12 DMU Rockets & 2-1-H Batteries | | 2199 |

3.3.1.1.2 Satellite Vehicle Reliability Budget Requirements

The reliability design objective for the satellite vehicle, excluding payloads, is 0.98 for the period of time during which the vehicle functions as a second stage booster in the ascent mode and 0.88 for subsequent operation in the active orbital mode for a period of 20 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 c omponent 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 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.

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

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

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

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

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

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

C. Provide a means for real-time commanding and stored program command- 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 by primary control and at least once by backup.

G. Provide the necessary orbit maintenance capability (Drag Makeup System - DMU) when flying shorter period and lower altitude orbits.

The Agena Model 39205, vehicle is a liquid-propellant second stage booster, powered by a gimballed rocket engine. During powered flight, pitch and yaw attitude control is provided by the rocket engine with roll attitude 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 section, the propellant tanks section, the aft equipment section, and the Stage II/Stage I adapter section.

The forward equipment section contains mounting provisions for the primary 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, hydraulic system, solar array, and secondary payloads. 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 7 depicts the satellite vehicle configuration.

Two types of propellants may be used by the rocket engine during its powered phase of the ascent trajectory. The first type consists of unsymmetrical dimethylhydrazine (UDMH) fuel and inhibited red fuming nitric acid (IRFNA) oxydizer. A nominal thrust of 16,100 pounds is achieved using the UDMH and IRFNA. The second type consists of unsymmetrical dimethylhydrazine mixed with Silicon oil (UDMH/SiO) fuel and high density acid (HDA) oxydizer. A nominal thrust of 16,870 pounds is achieved using the UDMH/SiO and HDA. 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 in conjunction with

P[redacted] VEHICLE

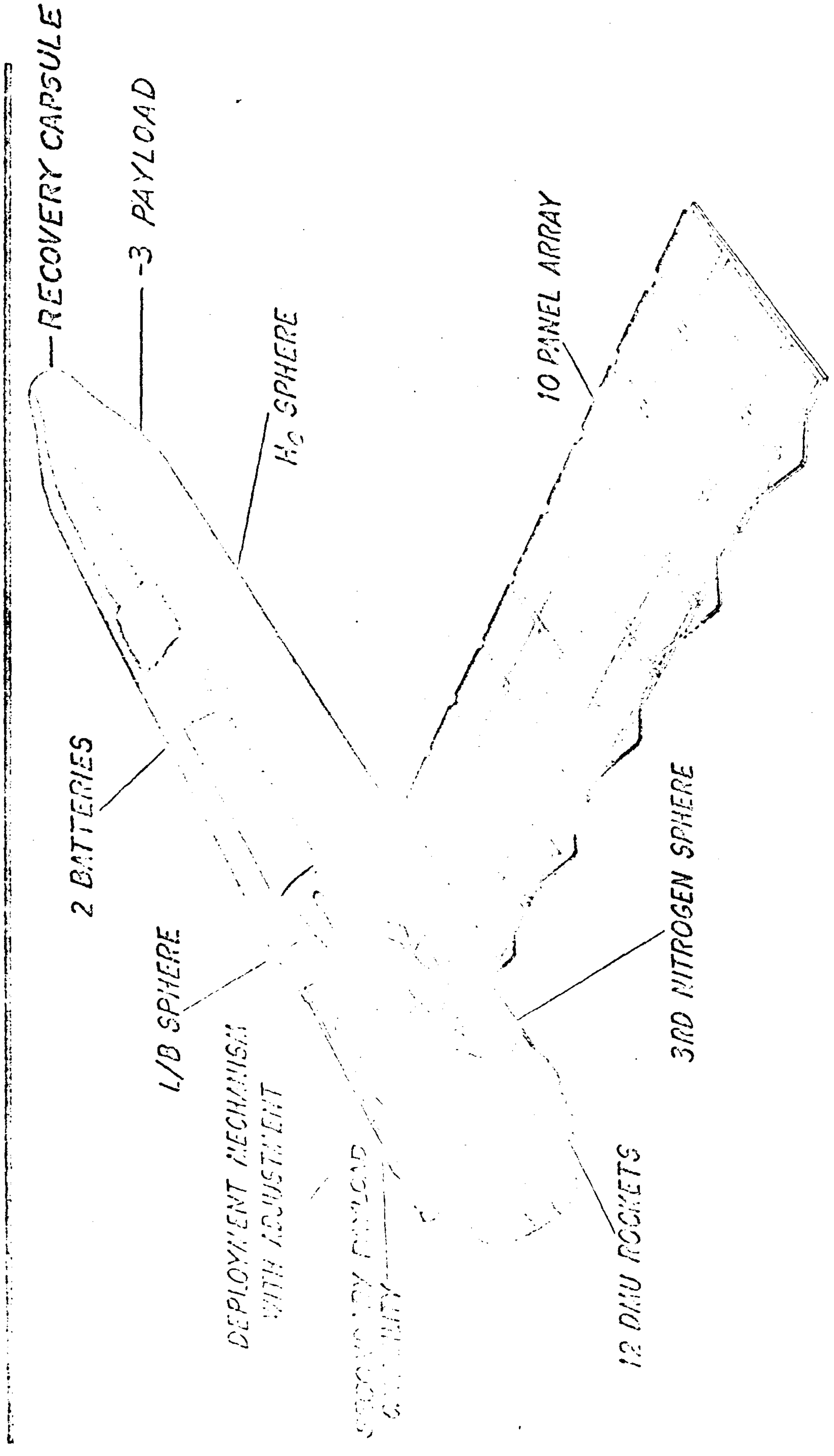



FIGURE 7 SATELLITE VEHICLE CONFIGURATION

the solar array 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 supplying three phase power and DC to DC converters operating from the unregulated DC bus to provide plus and minus regulated power.

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 gimballing 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 discrete command is employed to enable the velocity meter.

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 the SGLE uplink(SILO), 375 mc UHF (UNCLE), and Range Safety Command Links. The SGLE is an integrated tracking telemetry and command system. The system is integrated in the sense that all tracking and command data are multiplexed onto a single radio frequency carrier of 1.791 GHz for transmission to the vehicle. Similarly all telemetry and tracking data are multiplexed onto a single carrier of 2.237 GHz for transmission to the ground station. This carrier is coherently related to the ground-to-vehicle carrier. Link 2 is non-coherent and 5 MHz lower in frequency than Link 1 or the coherent carrier. Range is determined by measuring the phase shift (propagation delay) experienced by a pseudorandom binary ranging code. This code, generated in the SCF transmitter coder, phase-modulates the ground-to-vehicle carrier. The signal is subsequently demodulated in the vehicle transponder, filtered, and used to remodulate the vehicle-to-ground carrier without further processing. This carrier is manipulated by the ground receiving equipment to provide a delayed replica of the original transmitted signal. A local model of the ranging code is generated in the receiver coder and, by suitable delay, is made to coincide with the received code through the use of cross-correlation techniques. The required delay, with respect to the transmitted code, is equal to the round trip transmit time and, thus, is a measure of spacecraft range. SILO or UNCLE commands shall be used for selection of programmed payload and vehicle functions and the KIK SILO (secure) commands to enable the primary re-entry sequence and the early A to B transfer. A backup command capability shall be supplied by a 375 mc UHF (UNCLE) receiver in conjunction with a 39 command digital decoder.

Link I is used primarily to report vehicle and payload status and environmental data. Link 2 is used to telemeter backup 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.

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

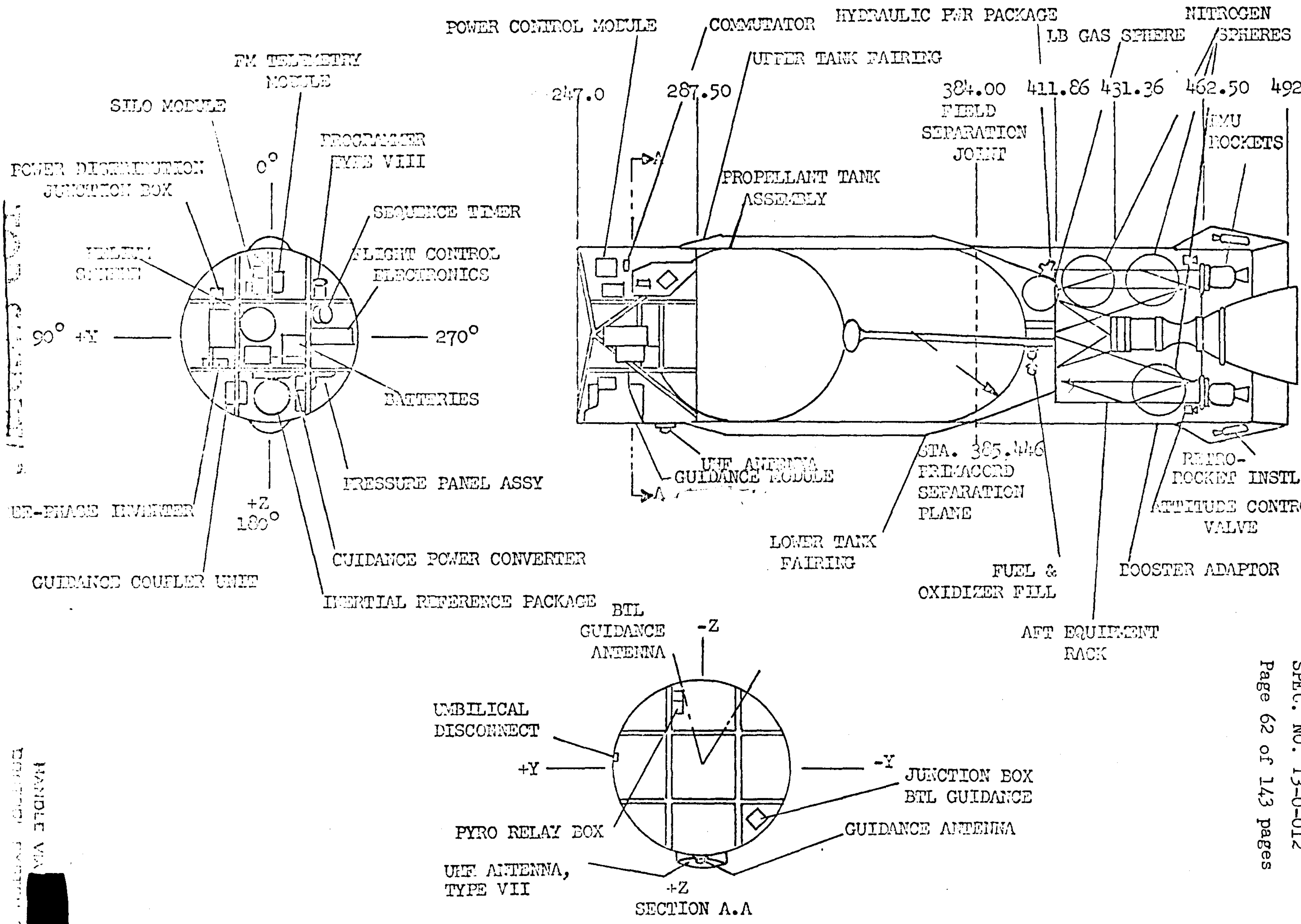



FIGURE 8. INBOARD PROFILE SATELLITE VEHICLE

during the mission. Each reel of tape shall accommodate 13 brushes to make electrical contact with an externally grounded drum through the punched holes providing a capability for 52 programmed commands. Tape speed shall nominally be 6.75 inches per subcycle and each tape length can be as long as 192 feet, providing programmed events for approximately 325 subcycles (orbit revolutions). Tape speed and positioning shall be adjustable by UNCLE or SILO 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 backup recovery timer shall be activated by an UNCLE secure command (KIK UNCLE) to control the Lifeboat backup recovery sequence.

3.3.1.2.2 Aerospace Ground Equipment (AGE)

The Satellite Vehicle Contractor shall provide the AGE required to checkout 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, counted down, and launched.

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. 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:

A. Assembly Buildings

B. Test Facilities

C. Clean Rooms

3.3.1.2.4 Satellite Vehicle Structural Envelope Requirements

Satellite vehicle equipment shall be contained within the structural envelope of the Agena. Figure 8 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)

3.3.1.3.1.1 Agena Command Subsystem

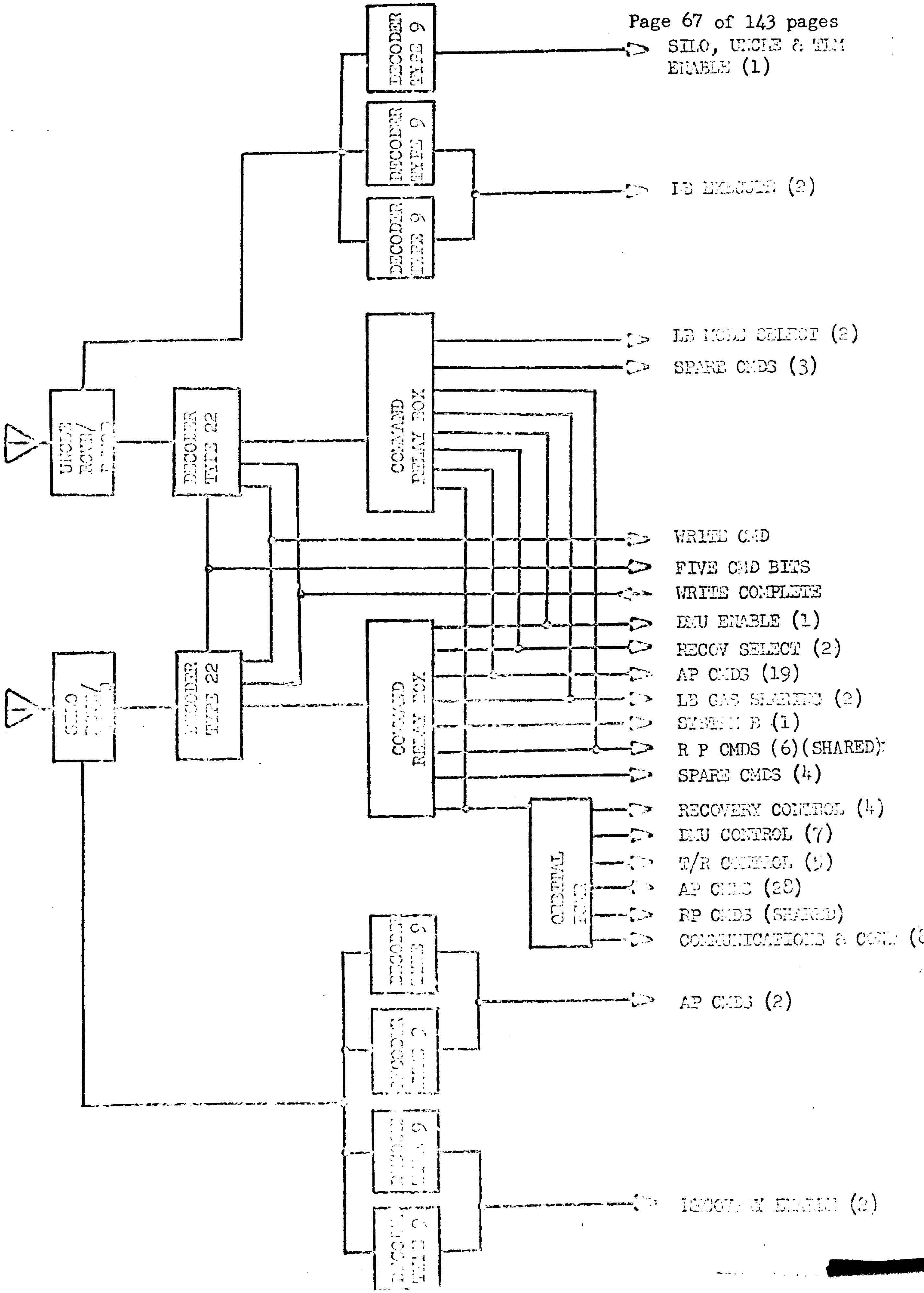
The satellite vehicle command subsystem shall provide real-time and stored commands for controlling all required events from powered flight through separation of the two re-entry vehicles. Critical functions shall be backed up in such a manner as to maximize assurance of successful commanding. A system 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 9.

A. Real Time Commands

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

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.


DECODER TYPE 9
DECODER TYPE 9
DECODER TYPE 9
ENABLE (1)
(2)
SPARE CMDS (3)
COMMAND RELAY BOX
DECODER TYPE 22
WRITE CMD
FIVE CMD BITS
WRITE COMPLETE
DMU ENABLE (1)
RECOV SELECT (2)
AP CMDS (19)
R P CMDS (6)(SHARED)
SPARE CMDS (4)
RECOVERY CONTROL (4)
DMU CONTROL (7)
ORBITAL
AP CMDS (2)

Command Destruct

The Stage I UHF Command destruct subsystem shall be equipped to meet or exceed the minimum requirements of SAMTEC as specified in AFWTRM 127-1 "Air Force Western Test Range Manual 127-1" 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 Tables 3 and 4.

1. Standard Timer - The Standard Timer shall provide capability for discrete timer interval commands with two to five functional events activated at each interval. Total running-time capability 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. Synchronization between programmed events and satellite position shall be maintained by adjusting the programmer period and/or by resetting the programmer tapes up to $\pm$ 150 seconds. Accuracy of the orbital programmer shall be plus or minus 4.5 seconds, including the effect of tolerances on tape punching and on-orbit adjustments.

TABLE 1

Real Time Commands

| | Digital (UHF) | | Secure | |
|---|---|---|---|---|
| | SILO | UNCLE | SILO | UNCLE |
| Orbital Programmer Increase/Decrease | X | X | | |
| Orbital Programmer Ten Second Step | X | X | | |
| Orbital Programmer Reset | X | X | | |
| Select Even Orbit Recovery | X | X | | |
| Select Odd Orbit Recovery | X | X | | |
| V/h Start Level | X | X | | |
| Orbital Programmer One Second Step | X | X | | |
| Lifeboat Next Orbit | | X | | |
| Primary Next Orbit | | X | | |
| L/B Control Gas Transfer | X | X | | |
| L/B Control Gas Transfer Stop | X | X | | |
| V/h Half Cycle Level | X | X | | |
| Emergency Camera Program Select | X | X | | |
| Emergency Intermix Operation Select | X | X | | |
| V/h Delay Start Position | X | X | | |
| Panoramic Camera Mode Select | X | X | | |
| Slitwidth Control Selector | X | X | | |
| DSR Load Disable & DSR Execute Enable | X | X | | |
| Drag Makeup System Enable | X | X | | |
| DISIC Camera Mode Select | X | X | | |
| Write Command (DSR Load) | X | X | | |
| Panoramic Camera No. 1 & No. 2 Exposure Control or Fail Safe | X | X | | |
| TLM Enable; Operational/Diagnostic Data Select | X | X | | |
| Panoramic Camera No. 1 Filter Change | X | X | | |
| Panoramic Camera No. 2 Filter Change | X | X | | |
| Exposure Control Delay | X | X | | |
| Yaw Programmer Enable/Disable | X | X | | |
| DISIC Camera Off/East/West/Both | X | X | | |
| Digital Storage Register Load Enable | X | X | | |
| Pressure Make-up Enable Disable | X | X | | |
| Pan Early A to B Switchover | | | X | |
| DISIC Early A to B Switchover | | | X | |
| Recovery Enable #1 | | | X | |
| Recovery Enable #2 | | | X | |
| Lifeboat Execute #1 | | | | X |
| Lifeboat Execute #2 | | | | X |
| Telemetry & RF Cmmd Links On (for 420 seconds) | | | | X |

TABLE 2

Radio Guidance Commands

Discretes

| | |
|---|---|
| Sequence 0 | No command |
| Sequence 1 | Main Engine Cutoff (MECO) - Stage I |
| Sequence 2 | Command Separation |
| Sequence 3 | Stage II Velocity Meter Enable |

Commands

| | |
|---|---|
| Pitch Up Steering Command | Booster and Agena |
| Pitch Down Steering Command | " " " |
| Yaw Right Steering Command | " " " |
| Yaw Left Steering Command | " " " |

TABLE 3

Stored Commands - Vehicle

| Function | Number of Commands | Command Mode |
| --- | --- | --- |
| **Ascent Phase** | | |
| Booster Separation Backup | 1 | Standard Timer |
| Enable Radio Guid. Steering | 2 | " " |
| Engine Ignition Sequence | 4 | " " |
| Flt. Control & Attitude Functions | 15 | " " |
| Disable Radio Guid. Steering | 4 | " " |
| Engine Shutdown Sequence (Backup) | 3 | " " |
| Switch TM to Orbit Mode | 2 | " " |
| Switch Flt. Control to Orbit Mode | 1 | " " |
| Arm Backup Rec. & Enable DMU Pyro Pwr | 1 | " " |
| Enable Primary Rec. Select | 2 | " " |
| Stop Ascent Timer | 2 | " " |
| **Orbit Phase** | | |
| Baseband & TM Link I On | 1 Brush | Orbital Prog. |
| Enable Tape Reset Command | 1 " | " " |
| Tape Index & Subcycle Identif. | 1 " | " " |
| Baseband & TM Link I Off | 2 " | " " |
| Disable Tape Reset Command | 1 " | " " |
| Re-entry Execute-odd Orbits | 2 " | " " |
| Re-entry Execute-even Orbits | 2 " | " " |
| TM Link II On | 1 " | " " |
| TM Link II Off | 1 " | " " |
| Pre-Launch Tape Index, Pneumatics to High Press | 1 " | " " |
| Start -36°/min Pitch Rate | 2 " | " " |
| Fire DMU Rocket | 2 " | " " |
| Disable DMU & Select Next Rocket | 2 " | " " |
| **Recovery Phase - Primary** | | |
| Vehicle Pitch-down Maneuver | 2 | Primary Recovery Timer |
| 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 |

TABLE 4

Stored Commands - Payload

| Function | Number of Commands | Command Mode |
|---|---|---|
| Ascent Phase | | |
| Switch Power for Door Eject (Inflight Reset) | 1 | Sep. Switches |
| Switch Power Backup for Camera, Cassettes, and Door Eject (Orbit Mode Signal) | 1 | Standard Timer |
| Orbit Phase | | |
| Emergency Program Intermix Advance, ECC Function Reset, Yaw Function Start, Oblateness Function Start | 1 | Brush(s) Orbital Prog. |
| ECC Delay Timer Start | 1 | " " " |
| Camera (ON-OFF) Sequence Programs 1 thru 9 | 16 | " " " |
| 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 | " " " |
| Emergency Program ON/OFF | 4 | " " " |
| Recovery Phase | | Both Primary and Lifeboat |
| Arm No. 1 Enable Capsule Recovery Events | 1 | Recovery Timer |
| Transfer Pwr to No. 1 Capsule T/C Battery | 1 | " " |
| Electrical Disconnect | 1 | " " |
| Fire No. 1 Capsule Eject Squibs | 1 | " " |
| Switch Over to B Mission | 1 | P/L Timer Function |
| Arm No. 2 Enable Capsule Recovery Events | 1 | Recovery Timer |
| Transfer Pwr. to No. 2 Capsule T/C Battery | 1 | " " |
| Electrical Disconnect | 1 | " " |
| Fire No. 2 Capsule Eject Squibs | 1 | " " |
| Recovery Enable (Capsule Batt Heater Turn On) | 1 | Secure Cmmd or L/B Timer |

3. Recovery and Lifeboat Timers

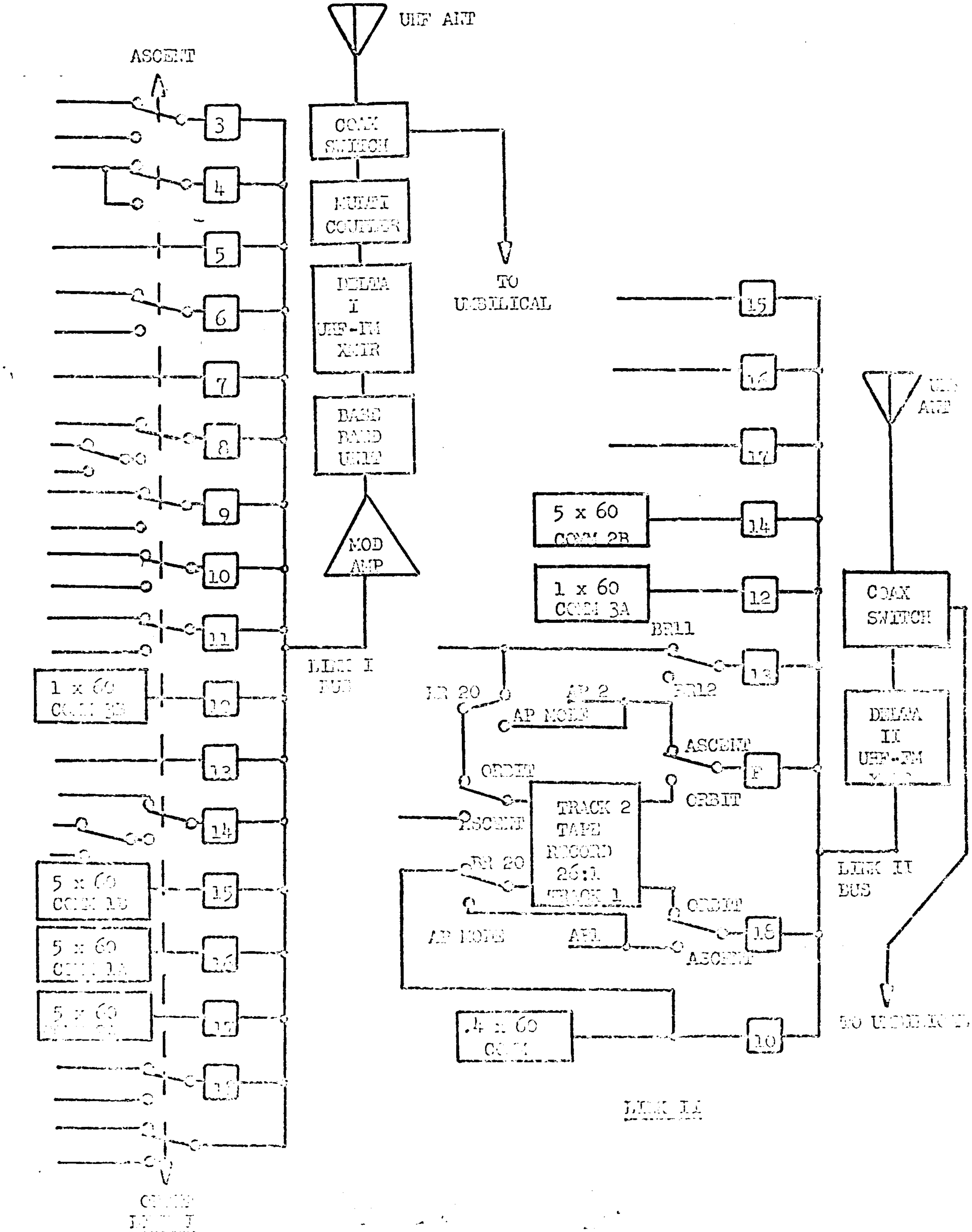
The recovery and lifeboat timers shall provide a capability to initiate at least 14 events (12 programmable and two fixed) each. A reset function shall be provided to reset the timer counter to the initial count and to reset the output relays. Normally, the reset pulse shall be generated by the timer 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 of the time 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. Transmissions in the 2200 to 2250 MC band shall be employed. A block diagram of the telemetry system is shown in Figure 10.

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.


UHF ANT
ASCENT
COAX SWITCH
DELTA I
BASE BAND UNIT
MOD AMP
TO UMBILICAL
LINK I BUS
5 x 60
1 x 60
COMM 3A
BR11
BR12
AP 2
ASCENT
ORBIT
TRACK 2
TAPE
26:1
COAX SWITCH
DELTA II
LINK II BUS
AP MODE
ASCENT
ORBIT
.4 x 60
3
4
5
6
7
8
9
10
11
12
13
14
15
16
17
18
F

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 one 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. Each error contributing element's maximum specified error shall be considered. All such error values shall be squared, the resulting squared values added together, and the square root of the resulting sum shall be taken to define overall error. Any error correction techniques shall not be considered in this definition of error. Maximum overall error for real time commutated analog data shall be ± 3%.

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

Under conditions of RF signal strength well above threshold, no 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 500 KHz/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.

D. Transmitters

Transmission shall be in the 2.2 to 2.3 GHz band and conform to IRIG requirements. Transmitter output power shall be a minimum of 2 watts (1.75 watts at the antenna terminal for Link II, 0.9 watts for Link I).

E. Subcarrier Oscillators

The subcarrier oscillators shall utilize standard IRIG bands with the exception of Channel F which deviates twice the specified range. The maximum subcarrier frequency drift as a result of all causes shall not exceed ± 2% of the bandwidth through which the subcarrier's frequency is deviated by full scale data. The subcarrier frequency deviation for a positive modulating voltage) with a linearity within ± 0.75%. Harmonic distortion shall not exceed ± 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 ± 1% of full scale. At least three calibration points shall be provided in each commutator 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 provided in the Satellite Control Facility.

H. Calibration Points

Each commutator shall contain calibration points for at least 0%, 50%, and 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. The calibration books shall contain curves or tables for each instrumentation points relating the magnitude of the physical quantity measured, in engineering terms, to the related subcarrier oscillator output. The subcarrier oscillator output shall be expressed both as a percentage of full scale and as an absolute frequency on the same sheet previously mentioned.

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

TABLE 5

Typical Assignment of Data Channels

| Channel No. | Mode | |
|---|---|---|
| Link I | Ascent | Orbit |
| 1 | Vehicle | Vehicle |
| 2 | Vehicle | Vehicle |
| 3 | Vehicle | Vehicle |
| 4 | Vehicle | Vehicle |
| 5 | Payload | Payload |
| 6 | Vehicle | Payload |
| 7 | Vehicle | Vehicle |
| 8 | Vehicle | Payload |
| 9 | Vehicle | Payload |
| 10 | Vehicle | Payload |
| 11 | Vehicle | Payload Commutated (0.4 x 60) |
| 12 | Vehicle Commutated (1 x 60) | Vehicle Commutated (1 x 60) |
| 13 | Payload Commutated (.4 x 60) | Payload Commutated (0.4 x 60) |
| 14 | Vehicle | Vehicle |
| 15 | Vehicle Commutated (5 x 60) | Vehicle Commutated (5 x 60) |
| 16 | Vehicle Commutated (5 x 60) | Vehicle Commutated (5 x 60) |
| 17 | Vehicle Commutated (5 x 60) | Vehicle Commutated (5 x 60) |
| 18 | Vehicle | Payload |
| Link 2 | | |
| 10 | Vehicle/Research Payloads | Vehicle/Research Payloads |
| 12 | Vehicle B/U Commutator (1 x 60) | Vehicle B/U Commutator (1 x 60) |
| 13 | [redacted] (0.4 x 60) | [redacted] (0.4 x 60) |
| 14 | Vehicle Commutated (5 x 60) | Vehicle Commutated (5 x 60) |
| 15 | Payload Commutated (5 x 60) | Payload Commutated (5 x 60) |
| 16 | Payload Commutated (5 x 60) | Payload Commutated (5 x 60) |
| 17 | Vehicle | Vehicle |
| 18 | Payload | Tape Recorder Track 1 |
| F | Payload | Tape Recorder Track 2 |

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

3.3.1.3.1.3 Tracking

The primary means of tracking the satellite vehicle will be the SGLS system operated by the Satellite Control Facility. The satellite vehicles shall contain an SGLE transponder and antenna compatible with the SCF tracking system. The transponder and antenna system shall provide a system margin of 10 db at 770 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 relability, 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 read-in 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 Hz, 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, Model 39205, equipment shall conform to the requirements of LMSC 6117B (or D), "General Environmental Specification for the Agena Satellite Program" except as specifically authorized. All primary payload 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 stablize 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 and DMU 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 Radioguidance Command shall initiate separation of Stage I from the satellite vehicle which starts the coast phase. Immediately following physical separation, the optical doors shall be ejected and the control of the satellite vehicle and rates about all three axes shall be initiated, utilizing the vehicle reference attained at VECO. The horizon sensors shall reference the roll axis to the earth horizon. Engine ignition shall be initiated by signal from the standard timer and engine shutdown 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 reacion 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 [redacted] missions are as given in Table 6.

TABLE 6

Dispersions at Orbit Injection

| Parameter | Requirement (3 sigma) | Objective (3 sigma) |
|---|---|---|
| Orbital Period* | ± 0.45 min. | ± 0.25 min. |
| Altitude of Perigee (from injection up to 65°N latitude) | ± 15 n.m. | ± 1.5 n.m. |
| Argument of Perigee | ± 80° (e > .008)<br>± 180° (e ≤ .008) | ± 5° (e > .008)<br>± 45° (e ≤ .008) |
| Inclination Angle | ± 0.30° | ± 0.25° |

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

B. Orbital Phase - Throughout the orbital mission, excluding de-boost sequence and drag makeup 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 may be subjected to the following momentum unbalances:

| Axis | Torque Impulse | Worst Case Mode |
|---|---|---|
| Pitch | 1.09 ft-lb seconds | Mono |
| Yaw | 4.04 ft-lb seconds | Stereo* |
| Roll | 10.5 ft-lb seconds | Mono |

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

The maximum restoring torque capability of the Guidance and Control system when in the low-gain orbit mode is:

| Axis | Torque |
|---|---|
| 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 (3 Sigma)

| 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 |

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 approximately six minutes.

C. De-Boost Phase

The de-boost sequence for the satellite vehicle is controlled by signals from a 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 62 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, Pitch-Down Attitude

| Function | Requirement (3 sigma) | Objective (3 sigma) |
|---|---|---|
| Pitch Angle from Local Horizontal ($120^\circ$/min $\pm$ 5% + $1.5^\circ$) | $7.5^\circ$ | $\pm 6.5^\circ$ |
| Yaw Angle from Orbit Plane | $2.0^\circ$ | $\pm 1.1^\circ$ |
| Roll Angle from Radius Vector | $1.5^\circ$ | $\pm 1.0^\circ$ |

In the event of a malfunction in the primary attitude control subsystem, a backup 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 and keeping the roll rate below $\pm$ 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 ability to return vehicle attitude control to the primary system subsequent to L/B operation, although not a requirement, presently exists.

D. Drag Makeup 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 are available in two sizes and have the following characteristics:

| | Low Impulse | High Impulse |
|---|---|---|
| Vacuum Thrust | 280 pounds | 408 pounds |
| Total Impulse - Vacuum | 2050 pound-second | 3075 pound-second |
| Nominal Delta Velocity/ Rocket | 15 feet/second | 24 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 | $I_z$ (Yaw) | $I_x$(Roll) |
| Separation from Stage I Booster | 18061 | 331.7 | -0.17 | 0.06 | 18,793 | 18,759 | 368 |
| Ignition Weight | 18060 | 331.7 | -0.17 | 0.06 | 18,793 | 18,759 | 368 |
| Burnout Weight | 4682 | 293.3 | -0.66 | 0.24 | 14,421 | 14,387 | 367 |
| Wt On-Orbit** | 4177 | 280.2 | 1.93 | 0.66 | 12,117 | 12,204 | 490 |

* Note: Does not include DMU rockets
** Solar Array deployed and no Secondary Payload

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 a solar array operating in conjunction with batteries to have sufficient capacity to supply electrical energy for all vehicle and payload requirements from liftoff, 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 a solar array and 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 20 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:

1. Three sigma low capacity of the battery (a 99.87 percent probability of meeting or exceeding the rated power capability).
2. Battery temperature environment and the consequent effect on usable battery capability.

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

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

5. Solar array charge current profile based on inclination angle, launch date, and time.

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 (T3-9-006). Estimated maximum payload power requirement is 800 watt hours per day with the following requirements: (19 day average)

1. Unregulated DC with an average load of 15 amps continuous, with 38 amp peak not to exceed 3 seconds (duty cycle of 20 minutes on and 70 minutes off).

2. 400 ± .008 cycles at 115 ± 2% volts AC with an average load on Phase C of 0.5 amps continuous and peaks of 0.75 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.

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

[redacted]

C. Electromagnetic Interference Control

The satellite vehicle shall be designed to comply with LMSC 447969B and the payload to LMSC T3-6-002 to the extent specified in paragraph 3.2.1.7.

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.

Fusistors shall be provided 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 for 500 millisec. 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 from 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 components or subsystems operation.

F. Grounding, Bonding, and Shielding

Grounding, bonding, and shielding of electrical subsystem components shall be accomplished in accordance with LMSC 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:

1. Pyrotechnics
2. Electrical Power
3. Commands
4. Telemetry Instrumentation
5. 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

F. Attachment and separation provisions

D. 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. Propellant tanks shall be provided to contain a nominal propellant load of 13,520 lbs. Propellants shall consist of two types. The first type consists of unsymmetrical dimethylhydrazine (UDMH) fuel and inhibited red fuming nitric acid (IRFNA) oxydyzer. The second type consists of unsymmetrical dimethylhydrazine mixed with silicon oil (UDMH/SiO) fuel and high density acid (HDA) oxidyzer.

For the [redacted] Mission, a single propulsive interval is normally required for the satellite vehicle, and restart of the rocket engine shall not be required. 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 degradation. 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 liftoff through orbit injection. However, if flight data should indicate that predicted dispersions are significantly conservative, the propellant contingency will be recalculated as required with prior approval of the [redacted] Program Directorate. Performance margin and propellant contingency shall be verified prior to each [redacted] vehicle launch. A [redacted] Program launch shall not be conducted under conditions that yield negative performance margin, or a propellant contingency corresponding to less than a 90 percent probability of achieving the desired orbit, unless directed to the contrary by the [redacted] Program Directorate.

3.3.1.3.7 Guidance Requirements - Ascent

The satellite vehicle shall contain the missile-borne equipment (MBGE) for steering of both the Stage I booster and the Satellite Vehicle during ascent flight. The MBGE shall include a radar transponder to aid ground tracking, a command receiver, and circuitry for utilizing the RF commands to control subsystem. Command signals shall be provided from the satellite vehicle across the interface to the Stage I booster in accordance with provisions of SLV-2H/Agena Electrical Interface Control Specification LMSC 1417293 and the Electrical Interface Drawing 1388789.

The function of the radio guidance system shall be to increase injection accuracy by providing real-time sequenced events and real-time steering corrections to the SLV-2H 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. The destruct charge ruptures propellant tank bulkhead, allowing mixture of fuel and oxidyzer, destroying satellite.

The Agena flight termination subsystem shall be equipped to meet the minimum command destruct requirements of SAMTEC 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-2H/Agena (LMSC 1417293) and the Electrical Interface Drawing 1388789.

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 1810 pounds. 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 image motion compensation (IMC) programmer voltage shall be directly proportional, selectable by commanding the IMC programmer, to match the velocity/height (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.

Programs for camera on-off operation shall be provided as in-flight loaded bits in a Digital Storage Register (DSR) and executed by stored commands from 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 de-boosted 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 subsystem 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-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 Contractor's 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 related to the [redacted] Program shall be identified and substantiated. Facilities shall be approved by the PRS/WCPO. 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 11 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 liftoff 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 liftoff through separation of SRV's. The satellite re-entry vehicles shall each contain beacon and telemetry subsystem 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 executing the DSR in-flight loaded commands shall be capable of selection through real-time commands to match the payload periods of operation and photographic equipment functions to actual orbital

PAYLOAD INBOARD PROFILE

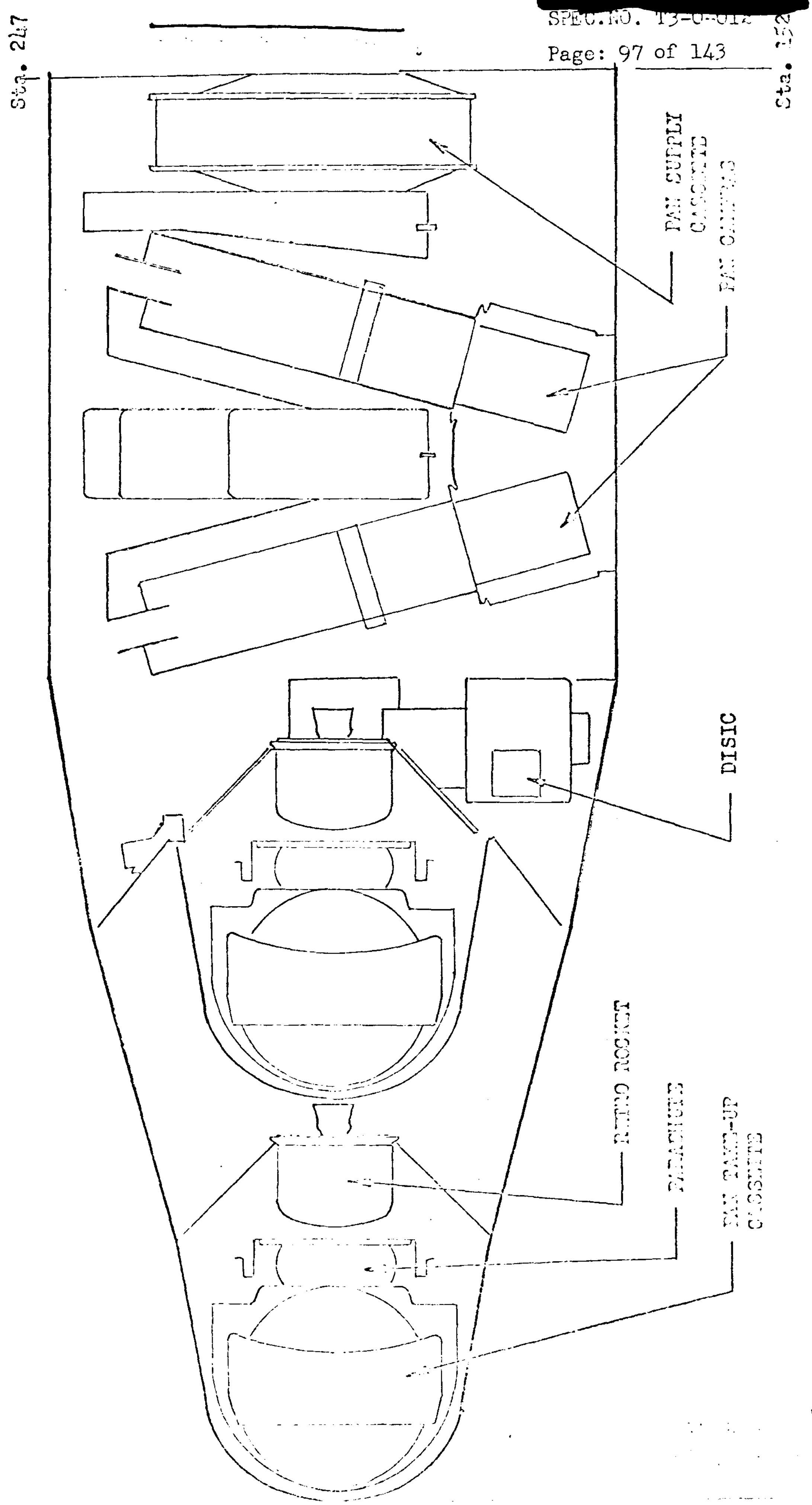



Figure 11

[redacted]

and ground track conditions during the mission. Command functions shall provide adequate flexibility to accommodate the combined effects of ascent flight dispersions (1 sigma) and variations in orbital parameters due to atmospheric drag and earth oblateness. Payload real-time command shall be accompained 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 (LMSC T3-9-006).

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 UHF 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 (LMSC T3-9-006).

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 degree of alignment accuracy required for camera optics and film transport, and for maintaining alignments under conditions of booster-induced environments during ascent and throughout the subsequent orbital phase. The payload section shall provide on-orbit environmental 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 a light-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 X $10^{-5}$ mm of Hg. or higher, as selected, 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 the 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 T3-9-006. The payload section shall provide power distribution networks for all payload equipment forward of the 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 J3 Payload section. For detailed description of this equipment refer to the applicable paragraphs of "J-3 Payload Section-Corona J Program" (RO-J3-001 or RO-J3-002).

A. Panoramic Camera

The panoramic cameras shall be capable of generating stereoscopic or monoscopic photographs of the ground at vehicle 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 of 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 24 inches. The lens shall be focused for orbital vacuum conditions. Each panoramic camera shall demonstrate a minimum dynamic resolution of 110 lines per millimeter using a USAF standard (2.1) test target and 100 percent match of image motion compensation.

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 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 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 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 2200 feet of terrain film and 2200 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. LMSC shall furnish a Control Sequence Box to provide control of the DISIC subsystem. LMSC 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 camera 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 11. The primary function of the SRV shall be the return of payload material from orbit. This shall be accomplished by separation of the SRV from the satellite vehicle, deboost from orbit, re-entry and subsequent parachute deployment and ablative heat shield separation. Recovery shall be effected by locating the descending capsule by means of recovery aids, and accomplishing aerial pickup by specially equipped aircraft. As a backup in the event air recovery is not successful, the capsule shall float and be acquired by a surface force.

The basic SRV shall be provided as government furnished equipment (GFE) to the Payload Contractor for installation of the take-up cassettes and parachutes. Attachment and separation provision, including command signals and instrumentation, shall be provided by the Payload Contractor. Figure 12 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 possible.

3.3.3.1.3 Facilities

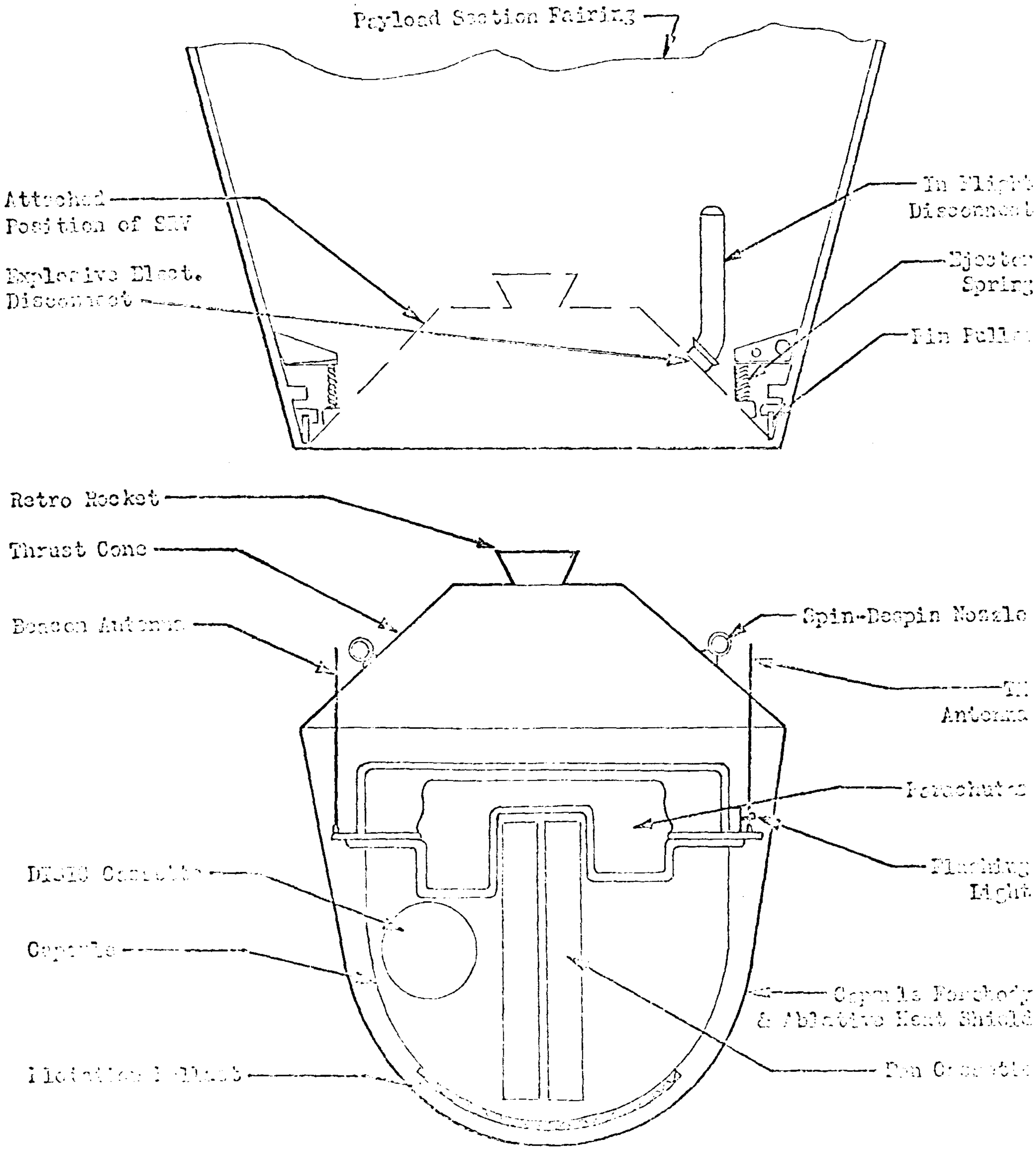
No special facilities other than clean rooms are expected to be required for the SRV. Any requirements of this type, uniquely related to Program [redacted] shall be identified and substantiated.

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

## SATELLITE RE-ENTRY VEHICLE INBOARD PROFILE



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 LMSC T3-6-002.

3.3.3.2.3 Attitude Control Requirements

During the separation sequence, the Satellite Vehicle shall provide 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 retro-rocket 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 <u>Structural Design Requirements</u>

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 descent at 10,000 ft above mean seal 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 ± 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 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, a VHF Beacon, and a 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 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-2H Thorad booster shall be as follows:

| Item | Weight (lbs) | Total Weight (lbs) |
|---|---|---|
| Weight Empty | | 7,797 |
| Propellants | 146,434 | |
| Pressurization Gas | 686 | |
| Solid Motor Boosters (3) | 29,589 | |
| Stage I Weight at Liftoff | | 184,506 |
| Less Expendables | 50,947 | |
| Weight at Solid Motor Burnout (38.6 sec) | | 133,559 |
| Less Expendables | 42,331 | |
| Less Solid Motor Cases (3) | 4,803 | |
| Weight at Solid Motor Separation (102 sec) | | 86,425 |
| Less Expendables | 77,211 | |
| Weight at Main Engine Cutoff (219.9 sec) | | 9,214 |
| Less Expendables | 163 | |
| Weight at Vernier Engine Cutoff (228.9 sec) | | 9,051 |

3.3.4.1.2 Launch Vehicle Reliability Budget Requirements

The reliability design objective for the launch vehicle is outside the scope of this specification. However, for the purpose of reliability apportionment, a figure of 0.97 is to be utilized. The SLV-2H 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 sub-orbital burnout condition is achieved within specified tolerances. Guidance commands shall be transmitted to the Stage I booster from a receiver located in the satellite vehicle.

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

E. Separate from the satellite vehicle at the required sub-orbital 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-2H Thrust Augmented Thorad is a vertically launched, liquid-fueled space booster powered by a main gimballed rocket engine and three thrust augmentation solid propellant rocket motors. Pitch and yaw control is provided by gimballing the main engine in the pitch and yaw planes during powered flight. Two gimballed vernier rocket engines provide roll control, and augment the main engine in providing pitch and yaw attitude control prior to main engine cutoff (MECO). Liquid propellants consist of RJ-1 for fuel and liquid oxygen.

The booster configuration is illustrated in 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 are attached externally to the sides of the booster structure and are jettisoned as early in the flight as range safety permits after their burnout has occurred some 40 seconds after liftoff. At the time of booster separation from the satellite vehicle, a retrovelocity 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 the satellite vehicle 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.

3.3.4.2.2 <u>Aerospace Ground Equipment (AGE)</u>

The Launch Vehicle Contractor shall provide the AGE required to checkout 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 subsystem 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 specification.

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 these items of equipment requiring confidence testing at limited time intervals, to maximize the time that the booster may be held on stand prior to recycling. After installation on the launch stand, the booster shall be checked-out for compatibility with launch AGE and subsequently with the mated satellite vehicle and radio guidance subsystem.

Solid rocket boost augmentation motors shall be attached before the satellite vehicle has been mated to the Stage I booster for launches from SLC-3W. At the backup launch pad, SLC-1E, solid rocket boost motor installation shall take place after the satellite vehicle has been mated to the 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
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, the booster shall be controlled by an autopilot flight controller. The flight controller shall maintain booster stability and shall direct the booster to the end of propulsion point as programmed for the flight. Radio guidance steering as vernier correction 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:

| Event | Time from Liftoff (sec) |
|---|---|
| Liftoff | 0 |
| Start roll to launch azimuth | 2 |
| Start pitch rate | $11.0^+$ |
| Stop roll rate | $11.0^-$ |
| Pitch rate step | 28 |
| Solid motor burnout | 37 |
| Pitch rate step | 83 |
| Drop Solid Motor Cases, Enable Radio Guidance | 102 |
| Pitch rate step | 104 |
| Start "Dog leg" yaw rate (if required) | 104 |
| Stop yaw rate, Start roll rate | 120 |
| Stop roll rate, Start Radio Guidance | 124 |
| Enable Main Engine Cutoff (MECO) | 200 |
| End Radio Guidance Steering | 216 |
| Main Engine Cutoff (Radio Command or Depletion) | 220 |
| Vernier Engine Cutoff, Stop pitch rate | 229 |
| Separate from Satellite Vehicle (Radio Command) | 243.5 |

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-2H/Agena Electrical Interface Specification.

3.3.4.3.2 Propulsion Requirements

The propulsion subsystem of the SLV-2H consists of a main liquid-propellant engine, two liquid-propellant vernier engines, and three solid propellant thrust augmentation rocket motors. Nominal performance characteristics for the propulsion subsystem shall be as shown in Table 10.

TABLE 10

Typical Booster Propulsion Performance Characteristics

| | |
|---|---|
| Liquid Engine | |
| Sea Level Thrust | 172,120 pounds |
| Total Impulse - Vacuum | 41,621,005 pound seconds |
| Solid Engines | |
| 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 |

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 ± 2 percent |
| Specific Impulse (SL min) | 248 sec |
| Propellant Utilization (min) | 99.8 percent |

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 <u>Electrical Power Requirements</u>

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

3.3.4.3.4 Flight Termination Subsystem

The range safety equipment installed in the booster vehicle shall conform to existing Western Test Range requirements. The installation shall consist of two command destruct receivers and two separate antennas. Each command destruct receiver shall be supplied with power independently of the other. The receiver outputs for destruct commands will be fed into the safety and arming mechanisms. A destruct command signal shall be provided to the Satellite Vehicle in accordance with the SLV-2H/Agena Electrical Interface Specification.

3.3.4.3.5 Telemetry and Tracking

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

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

3.3.5 Ascent Radio Guidance Subsystem

3.3.5.1 Description

The Western Electric Company Guidance System shall be used to provide real-time sequenced events and real-time steering corrections to the SLV-2H and Agena during the powered flight phase. The guidance equations utilize velocity steering to guide both the first and second stages. 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, 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.

Normal pre-flight preparations require 25 calendar days to generate necessary performance data, guidance computer tape, and check out the ground guidance equipment. 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.

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

3.3.6.1.1 Payload Camera/Satellite Re-entry Vehicle

An Interface Specification (T3-5-019) 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 (T3-5-020) 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 LMSC Drawing 1324217, while the electrical interface is defined by T3-9-006.

3.3.6.1.3 Satellite Vehicle/Satellite Control Facility

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

3.3.6.1.4 Satellite Vehicle - AGE

The electrical interface between the satellite vehicle (excluding the payload section) and AGE is defined in LMSC A068386. AGE requirements are established by the satellite vehicle system test specification and system design requirement directives.

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 physical interface between SLV-2H and Model 39205 vehicles is defined by LMSC Drawing 1386744. The electrical interface is defined on LMSC Drawing 1388789. The specification covering this interface is LMSC 1417293.

3.3.6.1.6 Satellite Vehicle/Ascent Guidance

The interface between the Satellite Vehicle and ascent guidance shall be accommodated by Specification for the airborne guidance equipment, the preflight performance data and trajectory reports, and the Program Requirement 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 receive 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 [redacted] Test Operations Order, and the Test Group Operations Plan.

3.3.7 Vulnerability Considerations

3.3.7.1 Modification 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] 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.

3.3.8 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 PRS/WCPO 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), AFSCF, 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 PRS/WCPO as appropriate. Support will also be supplied by SCF for operational tests, integrated systems test, and associated program activities which will allow performance of the program objectives in an efficient and reliable manner.

3.3.8.1 <u>Satellite Control Center</u>

The Satellite Test Center shall perform the following functions:

A. <u>Data Processing</u>

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 SGLS tracking equipment and provide tracking data to the STC for use in ephemeris determination.
3. Accept command data from the STC and provide the tracking 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.

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

B. Ephemeris Determination

Ephemeris 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 activities coincident with adequate program support. Ephemeris prediction capability must be very accurate and should have the capability of accounting for such factors as geopotential harmonics through the fourth order and a seventh parameter fit for average drag determination. Satellite spatial position errors on orbit must be known within the following limits:

1. ± 4.0 N.M. in track
2. ± 1.0 N.M. cross track
3. ± 0.5 N.M. in altitude

[redacted]

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 in order to select commands necessary to control the camera operations. Payload commands for the Digital Storage Register (DSR) will be generated using LETHAL software system, with manual backup provided by LMSC-AP.

D. Command Selection

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

3.3.8.2 Ground Stations

The basic SGLS is an integrated tracking, telemetry, and command system. The uplink data consists of a basic carrier with command and tracking words. The downlink carrier consist of telemetry and tracking words. [redacted] and [redacted] will use a 14 foot diameter parabolic reflector antenna while [redacted] and [redacted] will use the 60 foot diameter parabolic reflector and [redacted] and [redacted] may use a 14 foot antenna.

A. Tracking

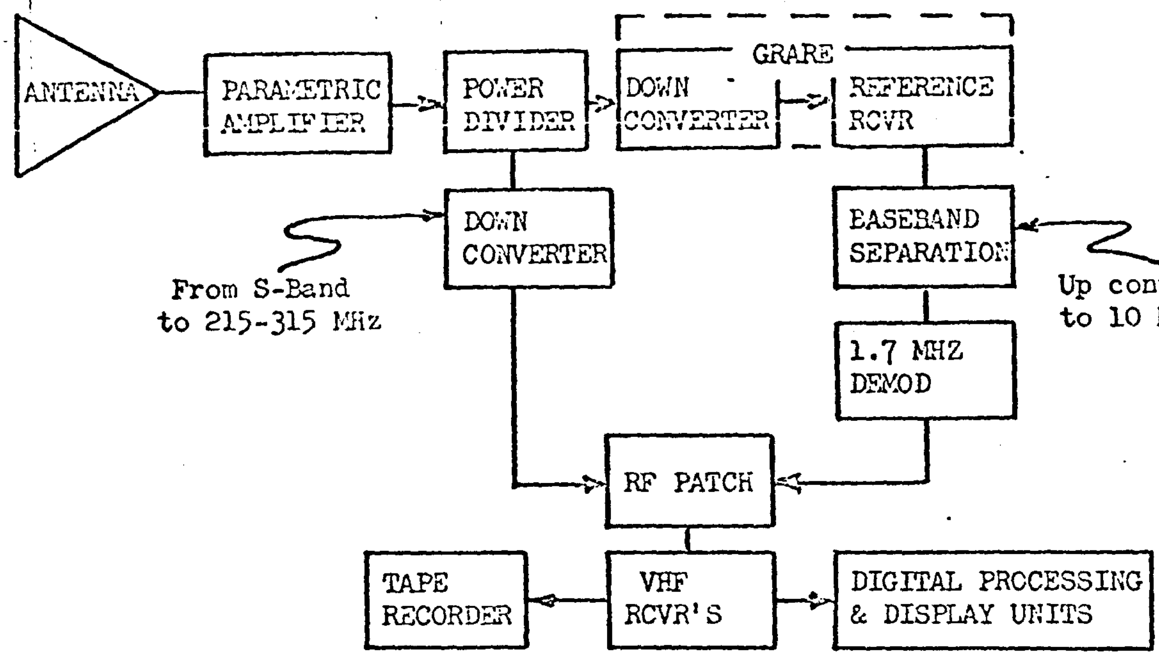
Range is determined by measuring the phase shift (propagation delay) experienced by a pseudorandom binary ranging code. This code, generated in the SCF transmitter coder, phase modulates the uplink carrier. The signal is subsequently demodulated in the Vehicle transponder, filtered, and used to remodulate the downlink carrier. The binary code is checked with the initial code and the difference or phase shift is a measure of Range. Doppler shift of the received signal is utilized to determine Range Rate. Range data and Range Rate data may be used separately or in conjunction for ephemeris determination.

During the orbital phase [redacted] and [redacted] will provide tracking support by acquiring and tracking on the SGLS link. Additional coverage can be provided by angle-tracking the telemetry links.

For the re-entry phases, vehicle tracking shall be provided by [redacted] and [redacted] utilizing the SGLS. [redacted] will track the satellite vehicle as it continues in orbit after SRV separation, and in addition, will receive angle-only data from the re-entry capsule VHF telemetry signal. The [redacted] station shall provide telemetry recording and data processing of the capsule telemetry only.

B. Telemetry

The basic SGLS equipment has been modified to process the Agena Link 2 telemetry. Only the A side of dual sided tracking stations has the capability of processing Link 2 data. The S-Band to VHF Down Converter as noted below is present only on the A side.




Carrier 1 is routed from the GRARE (Ground Receiving and Analog Ranging Equipment) to the Baseband Separation Unit. The output of the Baseband Separation unit is a 10 MHz signal which is sent thru the 1.7 MHz Demodulator to the 10 MHz input of the VHF receivers in the FM/FM Ground Station. Carrier 2 is routed from the SGLS antenna to the S-Band to VHF Down Converter. The output of the Down Converter is routed to the VHF receivers in the FM/FM Ground Station. Commutated and continuous data is then processed thru standard sub carrier discriminators, Channels 1 thru 18 and non-standard Channel F. Channel F uses a 98 KHz center frequency, with a deviation of ± 15%.

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 command systems for Program [redacted] are the SILO Command System, utilizing the SGLS uplink carrier and the UNCLE Command Link (UHF).

Each of the command links has a limited secure capability, referred to as KIK UNCLE and KIK SILO.

## 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-STD-785 and MIL-Q-9858A, as modified herein and by the authority of individual contract, shall apply. The payload shall comply to RO-J3-001 and RO-J3-002.

### 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.

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. 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 [redacted] 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 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 LMSC 6117B/D to the extent specified in the respective contractor's work statement. Payload equipment will be qualified to LMSC 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 baseline 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.

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 his end-item specification shown in the tree of Figure 6. Requirements at the functional subsystem level are as follows:

4.3.1.1 Satellite Vehicle

The Satellite Vehicle consisting of an Agena Model 39205 manufactured 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]rogram Directorate. Test results shall be presented in an acceptable data table format and be substantiated by detailed log books containing the test history, operations, equipment removals, failure data, and documented equipment configuration at the time of acceptance. Supporting analyses shall be presented to define any test anomalies and corresponding corrective action.

4.3.1.2 Launch Vehicle

The Stage I booster vehicle consisting of a standardized SLV-2H, 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 Agena. 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 Vehicle

The camera equipment and the Satellite Re-entry Vehicles shall be tested and accepted at the respective contractor facilities as shown in the flow diagram of Figure 4. Acceptance shall be by the procuring agency to the requirements of the acceptance specifications as shown in Figure 6. Detailed log books, test 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 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 STG. The countdown of the flight-ready vehicles shall be performed in accordance with a countdown manual approved by the Launch Base Test Group.

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 Space Test Group. The readiness review shall include but not be limited to the following participants:

A. Launch Base Test Group - 6595th STG

B. [redacted] Program Directorate

C. Photographic Reconaissance Systems/West Coast Project Office (PRS/WCPO)

D. [redacted]

E. SAMTEC

F. Air Force Satellite Control Facility

G. Ground Guidance Contractor - BTL/WECO

H. Stage I Booster Contractor - MDAC

I. Satellite Vehicle Contractor - LMSC

J. Payload Integrating Contractor - LMSC

K. Supporting Organizations for the Launch Wing, SAMTEC, and Satellite Control Wing, such as Weather, Range Safety, etc.

4.3.3 Acceptance Test Documentation

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

5.0 PREPARATION FOR DELIVERY

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

6.0 NOTES

For certain missions, a small excess performance capability is provided by the launch vehicle system 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 secondary payloads on the aft equipment rack of the Satellite Vehicle. Secondary payloads shall be functionally and electrically isolated from the Satellite Vehicle and its mission payload so that a malfunction of the secondary can in no way compromise the primary mission. Approval to incorporate secondary payloads on the Satellite Vehicle shall be contingent upon presentation of a thorough failure-mode analysis to substantiate the element of risk

[redacted]

involved to the [redacted] mission. Final approval to carry a secondary payload shall rest with the [redacted] Program Directorate. All secondary payloads to be carried shall be within the allowable weight margin for the mission and shall conform to the interface requirements specification for Agena Model 39205/Secondary Payloads (LMSC A817971) or individually issued Secondary Payload interface specifications.

## 6.1 List of Abbreviations

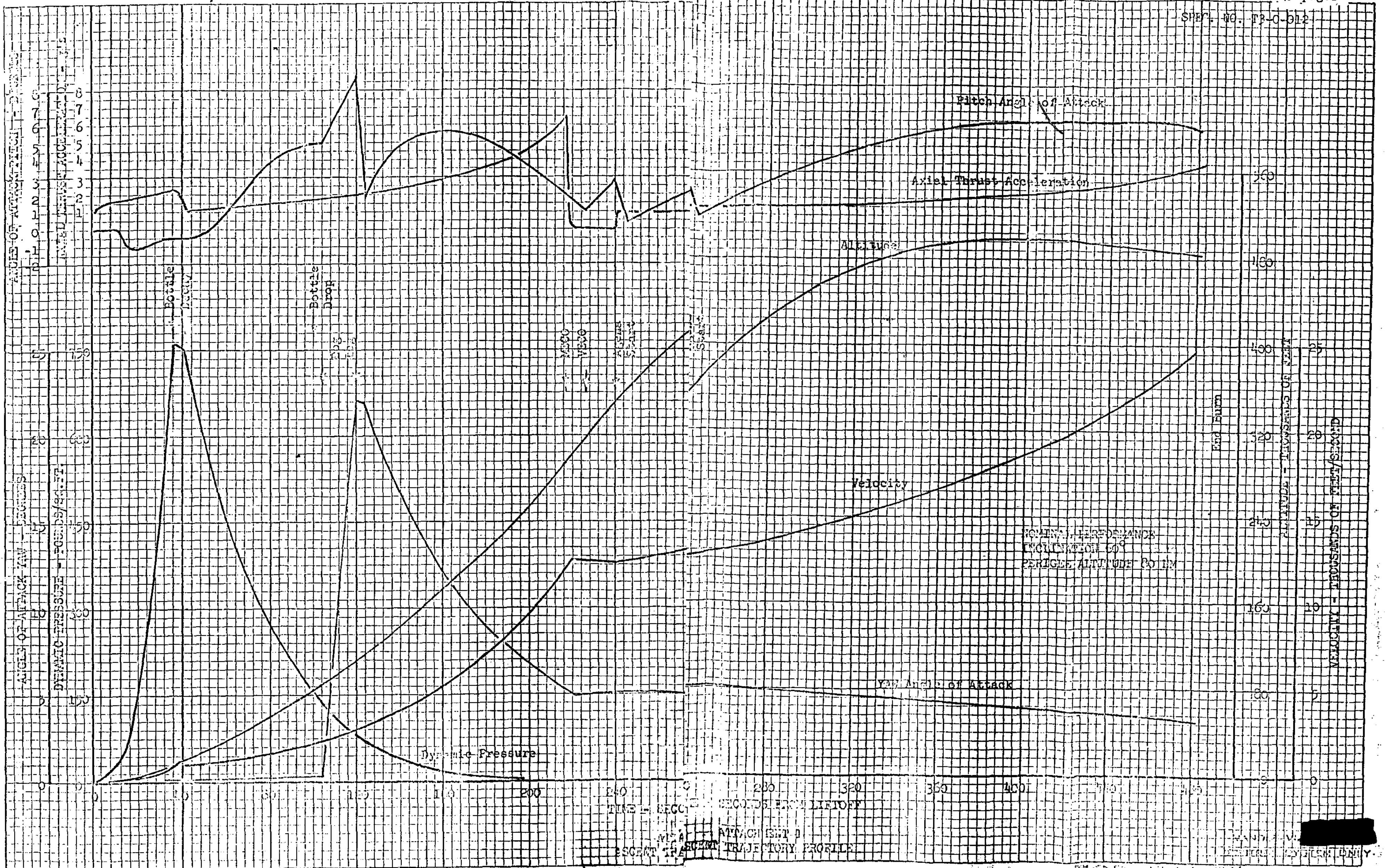
| Abbreviations | Full Terminology |
|---|---|
| AC | Alternating Current |
| AF | Air Force |
| AFSCF | Air Force Satellite Control Facility |
| AGC | Automatic Gain Control |
| AGE | Aerospace Ground Equipment |
| AP | Advanced Projects, Primary Payload |
| ARDC | Air Research Development Center |
| BTL | Bell Telephone Laboratories |
| BTU | British Thermal Unit |
| B/U | Backup |
| C&C | Communications and Control |
| Cg | Center of Gravity |
| Cmd | Command |
| CSE | Chief Systems Engineer (LMSC) |
| DB, db | Decibels |
| DC, dc | Direct Current |
| deg | Degrees |
| DISIC | Dual Improved Stellar Index Camera |
| DMU | Drag Makeup |
| DPE | Data Processing Equipment |
| DRCG | Digital Recording Clock Generator |
| DSR | Digital Storage Register |
| ECC | Eccentricity |
| EMI | Electromagnetic Interference |
| F | Fahrenheit |
| F/C | Flight Control |
| FCIC | Fairchild Camera and Instrument Corp |
| FM | Frequency Modulation |

| Abbreviations | Full Terminology |
|---|---|
| Flt. | Flight |
| Ft, ft | Feet |
| FTFD | Flight Test Field Director |
| G&C | Guidance and Control |
| GFE | Government Furnished Equipment |
| GRARE | Ground Receiving and Analog Ranging Equipment |
| GSETD | General Systems Engineering and Technical Direction |
| [redacted] | [redacted] Tracking Station |
| HG, hg | Mercury |
| HR, hr | Hour |
| H/S | Horizon Sensor |
| [redacted] | [redacted] Tracking Station |
| Hz | Hertz |
| IF | Intermediate Frequency |
| IMC | Image Motion Compensation |
| [redacted] | [redacted] Tracking Station |
| IR | Infra Red |
| IRIG | Inter-Range Instrumentation Group |
| IRP | Inertial Reference Package |
| KIAS | Knots Indicated Air Speed |
| [redacted] | [redacted] Tracking Station |
| L/B | Lifeboat |
| LV | Launch Vehicle |
| lb, lbs | Pound, Pounds |
| MBGE | Missile Borne Guidance Equipment |
| MECO | Main Engine Cutoff |
| MIL | Military |
| MM, mm | Millimeter |
| Min | Minute |
| MSL | Mean Sea Level |
| MDAC | McDonnell Douglas Astronautics, Western Division |
| N | North |
| [redacted] | [redacted] Tracking Station |
| NM, nm | Nautical Miles |
| [redacted] | North to South |
| [redacted] | [redacted] Tracking Station |
| ORD | Orbital Requirements Document |

| Abbreviation | Full Terminology |
|---|---|
| PACAFBASECOM | Pacific Air Forces Base Command |
| PAM | Pulse Amplitude Modulation |
| PDM | Pulse Duration Modulation |
| P/L | Payload |
| PM | Phase Modulation |
| PMU | Pressure Makeup |
| [redacted] | [redacted] Tracking Station |
| PPM | Pulse Position Modulation |
| PRD | Program Requirements Document |
| PRS/WCPO | Photographic Reconaissance Systems/West Coast Project Office |
| PU | Propellant Utilization |
| RCG | Recovery Control Group |
| R/E | Re-entry |
| Rec | Recovery |
| RF | Radio Frequency |
| RSC | Range Safety Command |
| RSS | Root Sum Square |
| RTC | Real Time Command |
| RPM | Revolutions Per Minute |
| SAMTEC | Space and Missile Test Center |
| S | South |
| [redacted] | [redacted] |
| SCD | Subcarrier Discriminator |
| SCF | Satellite Control Facility |
| SCO | Subcarrier Oscillator |
| SEC, sec | Second |
| SGLE (SGLS) | Space Ground Link Equipment (System) |
| SL | Sea Level |
| SLC | Satellite Launch Complex |
| SLV | Standard Launch Vehicle |
| SPC | Stored Program Command |
| SPD | Special Programs Director |
| SPL | Sound Pressure Level |
| [redacted] | [redacted] Auxiliary Tracking Station |
| SRV | Satellite Re-entry Vehicle |
| SSU | Sub-Satellite Unit |
| STC | Satellite Test Center |
| STD, std | Standard |
| STG | Space Test Group |
| STN | South to North |
| SV | Satellite Vehicle |
| TAT | Thrust Augmented Thor |
| TLM, T/M | Telemetry |
| TT&C | Telemetry, Tracking, and Command |
| TUMA | Cut and Splice Mechanism |
| T/C | Thrust Cone |

| Abbreviation | Full Terminology |
|---|---|
| UHF | Ultra High Frequency |
| USAF | United States Air Force |
| VAFB | Vandenberg Air Force Base |
| VECO | Vernier Engine Cutoff |
| VCO | Voltage Controlled Oscillator |
| VFG | Vehicle Function Generator |
| V/h | Velocity Height Ratio |
| VHF | Very High Frequency |
| V/M | Velocity Meter |
| [redacted] | [redacted] Tracking Station |
| W | West |
| WTR | Western Test Range |
| WH | Watt Hours |
| WECO | Western Electric Company |

[redacted]

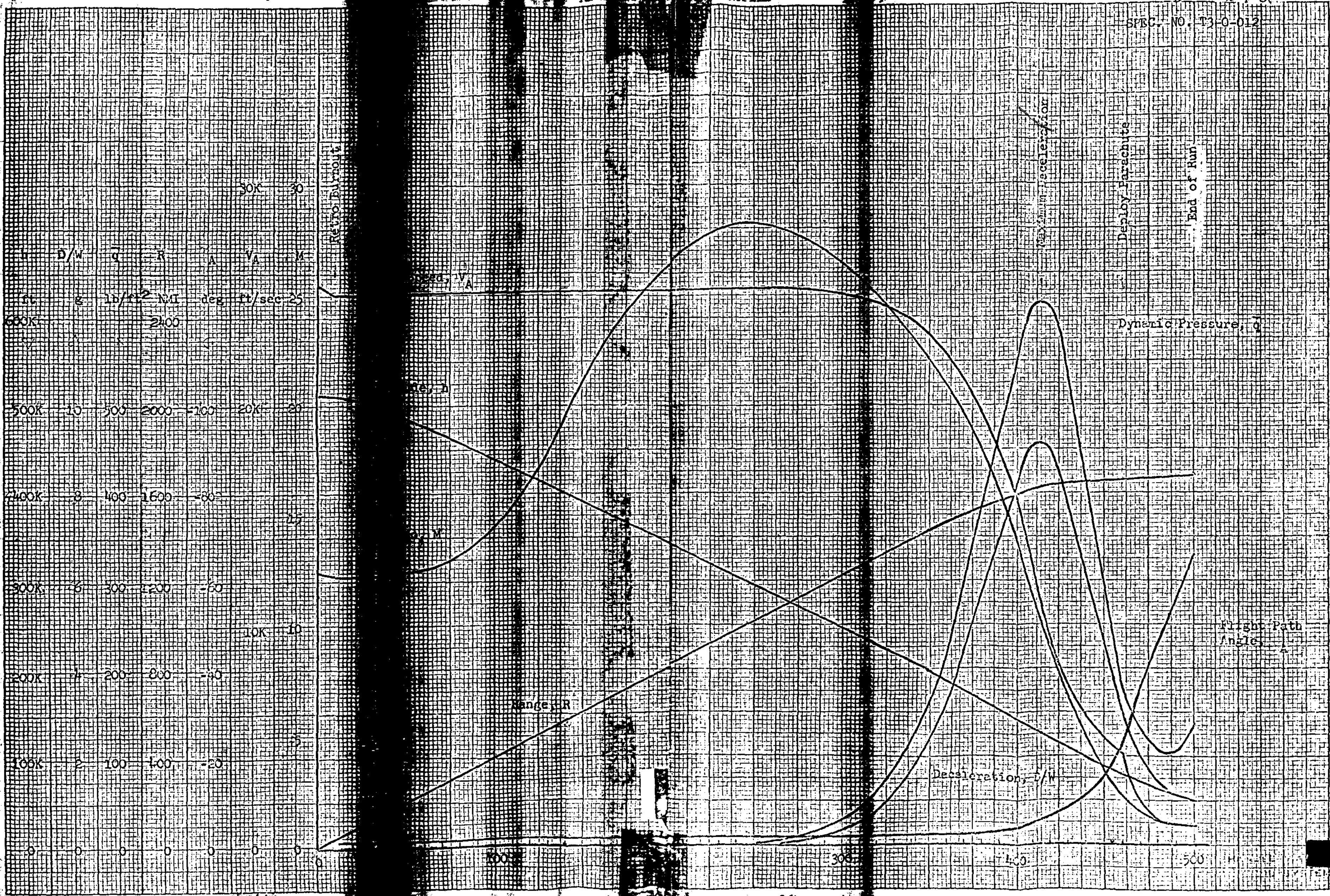




ATTACHMENT D
ASCENT TRAJECTORY PROFILE

[redacted]
CHANNELS ONLY

[redacted] ATTACHMENT 2

RE-E[redacted] PROFILE




[redacted]epresentative case 85 N Mi [redacted] 864.3 ft/sec retro velocity 321.5 lb SRV weight deboost angle-120 deg