8.2

Handle via [redacted] Control System

CORONA J-3 SYSTEM HANDBOOK

Handle via [redacted] Control System

- PREFACE -

This data book has been prepared by the National Reconnaissance Office with the assistance of the National Photographic Interpretation Center to provide a ready reference for operations and analysis.

Handle via [redacted]
Control System

CONTENTS

SECTION I

FORWARD

This Data Book provides an authorative, single-source reference of the Corona J-3/[redacted] Satellite Search-Surveillance System objectives, requirements, constraints, configuration data and engineering data. The term J-3 System encompasses the Thorad (SLV-2G) Booster, the Agena (SS-01B) Space Vehicle and the payload section including the panoramic cameras, DISIC, recovery and space structure subsystems. This document is based on current best information (base-line data and designs) and is intended to describe the system configuration as of July 1967. ~~There will undoubtedly be design modifications that will negate some of the information in this Data Book, therefore revisions of this publication will be issued as required to maintain its usefulness.~~

SECTION II

OBJECTIVE

The objective of the Corona J-3/[redacted] System is to obtain recoverable photographic reconnaissance data of specified geographical areas from a satellite vehicle. The fundamental purpose is to provide extensive stereoscopic coverage of the ground for intelligence acquisition. The secondary function is to provide photogrammetric control data as required for cartographic and geodetic compilations.

The J-3 System supersedes J-1. Significant design changes and improvements are enumerated as follows:

Parameter	J-1	J-3
Booster	Thrust-Augmented Thor	Thrust-Augmented Thorad
Active lifetime	11 days	14 days
Camera operating altitude	90-240 N.M.	80-200 N.M.
Perigee altitude	90-220 N.M.	80-110 N.M.
Ground resolution	10-12 ft.	7-8 ft.
Forward overlap	7.4%	7.6%
Weight on orbit		
Agena	2271	2428
P/L Section	1433	1754
	3404 lbs.	4182 lbs.
Mode of flight	Aft Forward	Nose Forward

Parameter	J-1		J-3	
Payload				
Diameter	50.5 inches		60 inches	
Panoramic Cameras				
Scan Function	360° rotating lens and oscillating scan head		360° rotating lens and scan head	
FMC	Translating Lens		Nodding Cameras	
Internal Calibration	None except PG flights		Panoramic Geometry	
Stellar Index Camera	Double frame camera		DISIC	
Units/System	2		1	
	Terrain	Stellar	Terrain	Stellar
Lens/Camera	1	1	1	2
Field of View	70°	16°	74°	23.5°
Focal Length	38 mm.	85 mm.	3 inch	3 inch
Film Loading/ Camera	400 frames	400 frames	4800 frames	16,000 frames
Horizon Cameras				
Filter	Wratten 25		Wratten 25 plus commandable attenuator	
Fiducials	4		4 plus one for reference	

SECTION III

REQUIREMENTS

The Corona J-3/[redacted] System is capable of obtaining stereoscopic ~~and monoscopic~~ photography by using panoramic cameras operating in orbit. The panoramic camera photographic scan angle is 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 6.2 million square nautical miles. The panoramic camera subsystem is capable of being programmed for the desired portion of the ground track on any given orbit. Supplementary provisions are provided to locate the vehicle position at exposure within one quarter minute of arc at the local horizon in relation to concentric earth coordinates, with a corresponding time determination within one millisecond.

A Dual Stellar Index Camera (DISIC) is used to obtain terrain and stellar photography. This camera is capable of running in a slave mode to the panoramic cameras or on an independent basis. The J-3 System is capable of performing missions of 14 days duration with early call-down capability. Dual recovery vehicles each having a capacity for one half the total film load of both the panoramic and DISIC cameras are used in the nominal mission. Recovery of the first capsule usually precedes continuation of the second half of the mission; however, should operational factors dictate, the film take-up function can be switched to the second capsule on ground command [redacted] the mission continued prior to recovery of the first capsule.

A Thrust-Augmented Thorad Boost Vehicle (SLV-2G) and a modified standard Agena (SS-01B) are used to launch from Vandenberg AFB (VAFB) SLC-1-E and SLC-3-W. In addition to performing the function of second stage boost vehicle, the Agena also serves as a stabilized satellite platform for support of the photographic mission and re-entry vehicles. The reentry vehicles containing the photographic record will be subject to air retrieval over water, or alternatively to water retrieval. On-orbit control will be performed using the Satellite Control Facility (SCF).

A general summary of the System flight parameters is tabulated below.

J-3 FLIGHT PARAMETERS

Parameter	Design Range	Nominal Range
Active lifetime	14 days	
Camera operating altitude	80-200 N.M.	80-120 N.M.
Period	88-91.5 Min.	
Perigee altitude	80-110 N.M.	
Inclination	60°-110°	80°-97°
Location of perigee	20°-60° No. Lat. Descending	
Beta angles	+65° to -65°	
Reentry time-first capsule	1 to 10 days	6 to 7 days
Reentry time-second capsule	2 to 14 days	14 days

The spatial position of the camera station can be determined to an accuracy of 1200 feet intrack, 600 feet crosstrack, and 600 feet in altitude.

SECTION IV

SYSTEM DESCRIPTION

Figure 1 illustrates the launch configuration for the Corona J-3/[redacted] System.

LAUNCH VEHICLE

The launch vehicle consists of a first stage SLV-2G Thrust-Augmented Thorad booster, a modified SS-01B Agena satellite vehicle functioning in an ascent mode as the second stage booster, and WECO/BTL radio guidance equipment for tracking and steering the booster vehicles into the desired orbit.

The SLV-2G Thrust-Augmented Thorad Booster

The Stage I booster vehicle performs the following functions during the ascent phase of the [redacted] mission.

A. Provides 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. Performs pre-programmed maneuvers to orient the launch vehicle configuration to the desired flight azimuth, maintains heading within range safety boundaries, and executes yaw maneuvers when required to achieve the azimuth necessary for particular orbits.

C. Maintains attitude control and responds 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.

 [redacted]

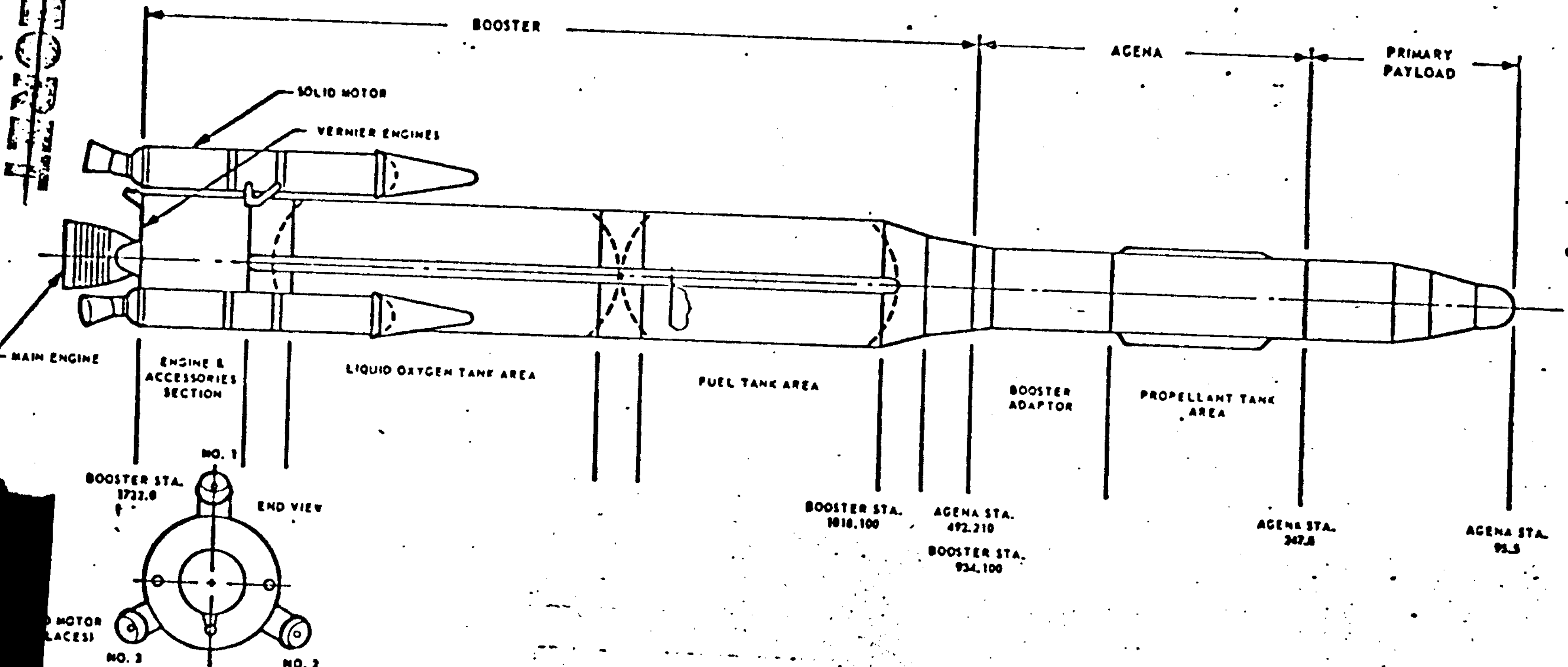


FIGURE 1 LAUNCH VEHICLE CONFIGURATION CORONA J3/[redacted] SYSTEM

 [redacted]

D. Provides tracking signals during ascent for range safety impact calculations and is 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. Separates from the satellite vehicle at the required sub-orbital flight condition by means of a retro-velocity maneuver without inducing rotational torques in the satellite vehicle.

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

The SLV-2G Thrust-Augmented Thorad is a vertically launched, liquid-fueled space booster powered by a main gimballed rocket engine and three thrust 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 by Figure 1, and consists of six structural sections. From forward to aft, the sections are designated: transition section, adapter section, fuel tank, center body section, oxidi er tank, and engine/accessories section. The solid propellant motors attached externally to the sides of the booster structure, are jettisoned early in the flight after their burnout has occurred some 40 seconds after liftoff.

At the time of booster separation from the satellite vehicle, a retro-velocity is imparted to the booster by solid rockets attached to the Stage I/satellite vehicle adapter. The adapter remains attached to the Stage I booster throughout separation, and carries with it the satellite vehicle's range safety destruct charge.

Weight Budget

The nominal weights for the SLV-2G Thorad booster are:

Item	Weight (lbs)	Total Weight (lbs)
Weight Empty		7,795
Propellants	145,356	
Pressurization gas	794	
Solid Motor Boosters (3)	29,538	
Stage I Weight at Liftoff		183,483
Less Expendables	92,087	
Less Solid Motor Cases (3) (Burnout at 40 sec.)	4,809	
Weight at Solid Motor Separation (100 sec.)		86,587
Less Expendables	77,095	
Weight at Main Engine Cutoff (218.4 sec.)		9,492
Less Expendables	193	
Weight at Vernier Engine Cutoff (227.4 sec.)		9,299

Guidance and Control

From liftoff until initiation of radio command guidance, the booster is controlled by an autopilot flight controller. The flight controller maintains booster stability and directs the booster to the guidance initiation point as programmed for the flight. Programmed maneuvers are implemented by a punched tape programmer/timer to actuate various portions of the control circuits. Subsequent to completing programmed orientation maneuvers, the guidance relay is locked in and the booster responds to guidance command steering adjustments provided to the flight controller from the receiver located in the satellite vehicle. Radio guidance steering is enabled by the flight controller and is terminated for booster steering by the ground guidance equipment just prior to booster main engine cutoff (MECO). MECO and satellite vehicle separation are commanded by radio guidance. All ascent guidance functions, with the exception that MECO occurs through propellant depletion, are backed up by a nominal flight program of stored commands in the event of radio guidance failure. With radio guidance, an accuracy of one percent of the computed radio guidance steering commands is achieved. If radio guidance is lost, the booster flight controller guides the booster to the burnout condition within the operational tolerances of the equipment. These are: $\pm$ 5 degrees in flight path angle, 9.6 nautical miles in position, and 450 feet per second in velocity.

Propulsion

The propulsion subsystem of the SLV-2G consists of a main liquid propellant engine, two liquid propellant vernier engines, and three solid propellant thrust augmentation rocket motors. Nominal performance characteristics for the propulsion subsystem are:

Liquid Engine	
Sea Level Thrust	172,120 pounds
Total Impulse - Vacuum	41,621,055 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 are:

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)	99.6 percent

Characteristics of each of the thrust augmentation solid motors are:

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.

Electric Power

The booster electrical power subsystem provides a source of A.C. and D.C. power required by the various booster vehicle components and equipment; however, the booster does not supply electrical power across the interface to the satellite vehicle.

Flight Termination

The range safety equipment installed in the booster vehicle consists of two command destruct receivers and two separate antennas. Each command destruct receiver is supplied with power independently of the other. The receiver outputs for destruct commands are fed into the safety and arming mechanisms. A destruct command signal is also provided to the satellite vehicle through the interface.

Telemetry and Tracking

The booster vehicle PDM/FM/FM telemetry subsystem provides a telemetered data for post-flight analysis of booster performance, environments, and sequenced events. Diagnostic data, suitable for analysis of booster malfunctions, is also obtained. The booster does not require a separate beacon for tracking purposes.

SS-01B Agena Satellite Vehicle

The satellite vehicle performs both ascent and orbital functions. The ascent functions are to:

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

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

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

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

The Agena SS-01B vehicle is a liquid-fueled second stage booster, powered by a gimballed rocket engine. During powered flight, pitch and yaw control is provided by the rocket engine with roll control provided by cold gas reaction jets. During coast and orbital flight, attitude control is effected by three-axis pneumatic reaction nozzles. The vehicle as illustrated in Figure 2 is composed of four major sections: the forward equipment section, the propellant tanks section, the aft equipment section, and the Stage II/Stage I adaptor section.

TOP SECRET [redacted]

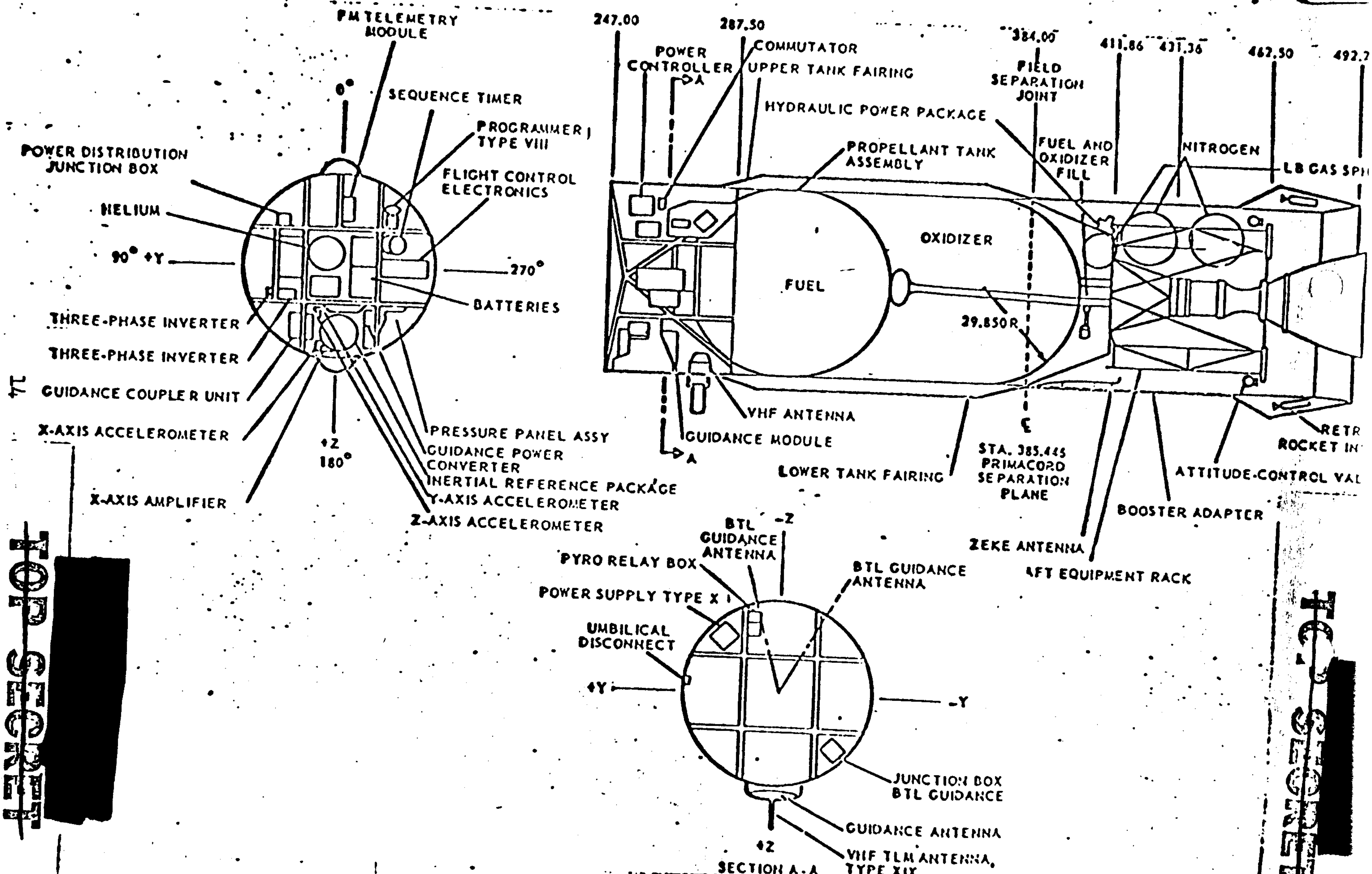


FIGURE 2 INBOARD PROFILE AGENA VEHICLE

TOP SECRET [redacted]

The forward equipment section contains the mating ring mounting provisions for the 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, seven DMU rockets, and hydraulic system. The booster adaptor section attaches to the aft part of the tank section and is designed to support the entire satellite vehicle from the first stage booster during the ascent phase. The adaptor section remains attached to the Stage I booster at the time the two vehicles are separated in flight.

Weight Budget

Nominal weights for the SS-01B Agena vehicle are:

Item	Weight (lbs)	Total Weight (lbs)
Weight Empty		1899
Propellants	13520	
Helium	1	
Attitude Control Gas (-3 Mix)	68	
Aux Att Control Gas -L/B (-3 Mix)	14	
5 IH Batteries	630	
7 DMJ Rockets	140	
Gross Weight Without Payloads		16272
Less Adapter and Attach	354	
Less Retro Rockets	10	
Less Destruct System	6	
Less Horizon Sensor Fairings	7	
Less Attitude Control Gas	1	

Item	Weight (lbs)	Total Weight (lbs)
Ignition Weight w/o Payloads		15894
Less Propellants	13422	
Less Engine Start Change	1	
Less Attitude Control Gas	3	
Burnout Weight		2468
Less Residual Propellants	48	
Less Helium	1	
Propellant Margin	50	
Weight on Orbit with Gas but w/o Payload		2369
Less Remaining Att Control Gas	64	
Less Remaining L/B Gas	14	
Empty Weight on Orbit w/o Gas, w/o Payload, W 7 DMU		2291
Rockets &W/ 5 IH Batteries		

Ascent Commands

Real-time commands are transmitted to the vehicle by X-Band radar and the Range Safety Command links. A continuous tracking X-Band radar which pulse-position modulates the command spacing between continuous pairs of address pulses, commands the BTL/radio guidance subsystem. The commands used are as follows:

	Discretes
Sequence 0	No command
" 1	Main Engine Cut Off (MECO) - Stage I
" 2	Command Separation
" 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	" "

Ascent Guidance and Control

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

The satellite vehicle contains the vehicle-borne radio guidance group for command steering of both the stage booster and the satellite vehicle during ascent flight. The vehicle-borne guidance group includes a radar transponder to aid ground tracking, a command receiver, and circuitry for utilizing the RF commands to control the necessary functions of the satellite vehicle guidance and control subsystem. Command signals are provided from the satellite vehicle across the interface to the Stage I booster.

The function of the radio guidance system is to increase guidance accuracy by providing real-time sequenced events and real-time steering corrections to the Thorad and Agena vehicles during their boost phases. Steering corrections, implemented by the radio guidance system, are in the nature of vernier corrections and, if none are received, the vehicle guidance and attitude control subsystem 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 are actuated by programmed stored commands if they are not commanded by radio guidance. Thorad MECO and Agena velocity meter enable commands are not backed up by a programmed command because engine shutdowns will occur upon propellant depletion.

At termination of Stage I booster thrust by guidance discrete command, the satellite vehicle standard timer begins operation. Subsequently, at the time of Stage I vernier engine cutoff, the inertial reference gyros in the satellite vehicle are uncaged and the horizon sensor fairings ejected. A coast phase is initiated upon separation of Stage I from the satellite vehicle by radio guidance, ~~discrete~~. Immediately following physical separation the optical doors are ejected and the control of the satellite vehicle attitude and rates about all three axes are initiated, using the vehicle reference attained at MECO. The horizon sensors reference the roll axis to the earth horizon. Engine ignition is 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 is enabled by a radio guidance discrete command. During the burn period, pitch and yaw control is provided by hydraulic actuation of the gimballed engine while roll control is maintained by pneumatic reaction control jets. Radio guidance commanding is used throughout the major portion of the burn period to provide pitch and yaw steering commands to the satellite vehicle. Orbital injection accuracies are as follows:

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

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

Ascent Propulsion

The satellite vehicle propulsion system provides thrust for second stage boost during ascent into orbit. The propulsion system, consisting of a liquid rocket engine and components, develops a nominal vacuum thrust of 16,000 lbs. The engine is designed for a nominal thrust duration of 245 seconds. The rocket engine thrust chamber is mounted on a gimbal ring and provides partial satellite vehicle attitude control during engine operations by means of yaw and pitch thrust chamber movement. Cylindrical shaped propellant tanks contain a nominal propellant load of 13,520 lbs. Propellants consist of Inhibited Red Fuming Nitric Acid (IRFNA) as an oxidizer, the Unsymmetrical Dimethylhydrazine (UDMH) as a fuel.

A single propulsive interval is used for second stage for the boost; restart of the rocket engine is not required. Normally a propellant contingency is reserved in the satellite vehicle to cover the root-sum-squared (rss) effect of minus 3 sigma dispersions. If the Stage I booster is utilized essentially to propellant depletion, the contingency carried in the satellite vehicle provides for RSS dispersions in flight from liftoff through orbit injection. Performance margin and propellant contingency are verified to insure against a negative performance margin, or a propellant contingency corresponding to less than a 90 percent probability of achieving the desired orbit prior to each launch.

Ascent Electric Power

An electrical subsystem provides power to operate vehicle and payload electrical equipment. Batteries are used as the primary power source for the 24 volt direct current supply. Power conversion is accomplished by a three phase inverter(s) supplying single phase power from one leg, and D.C. to D.C. converters operating from the unregulated D.C. buss.

Ascent Flight Termination

The satellite vehicle flight termination subsystem is 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 are carried in the Stage I booster. The satellite vehicle provides the capability for carrying two redundant sets of destruct signals across the interface from Stage I. Additionally, the satellite vehicle provides a destruct pyrotechnic charge located in the booster adapter, 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.

Ascent Radio Guidance

The Western Electric Company/BTL Guidance System is used to provide real-time sequenced events and real-time steering corrections to the SLV-2G and SS-01B during the powered flight phase. The guidance equations use velocity steering to guide both the first and second stages of powered flight. Controlled parameters for the first stage are apogee velocity, apogee radius and inclination. The Stage I booster engine is shut down by radio guidance command, as is separation of the Stage I booster from the Agena SS-01B 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 cause attitude changes of the vehicle during powered flight to implement thrust vector corrections. These corrections 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 does not force the vehicle to fly a nominal flight path, but commands steering corrections to assure a specified velocity vector at cutoff.

Normal pre-flight preparations require 8 days to generate necessary performance data, guidance computer tape, and check out the ground guidance equipment.

THE SATELLITE VEHICLE

The satellite vehicle consists of a modified SS-01B Agena satellite vehicle with a payload section and two modified Mark 5A Satellite Reentry Vehicles (SRV) attached.

SS-01B Agena Satellite Vehicle

The on-orbit functions of the Agena vehicle are to:

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

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

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

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

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

F. Perform necessary maneuvers and sequences to eject the two recoverable reentry vehicles from the satellite vehicle for de-boost from orbit by primary control and at least once by back-up.

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

Figure 2 illustrates the SS-01B vehicle configuration.

Mass Properties

Nominal weight estimates of the Agena vehicle appear in the description of the launch vehicle. Estimated Mass Properties excluding 5 DMU rockets are:

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

Tracking, Telemetry and Command

The tracking, telemetry and command subsystem consists of vehicle-borne transmitters, receivers, decoders and programmers.

The satellite vehicle command subsystem provides real-time and stored commands for controlling all required events from powered flight through separation of the two reentry vehicles. Critical functions are backed up in such a manner as to maximize assurance of successful commanding. A system of command interlocks are provided to minimize the effects of inadvertent command or covert interference. A block diagram of the command subsystem is included as Figure 3.

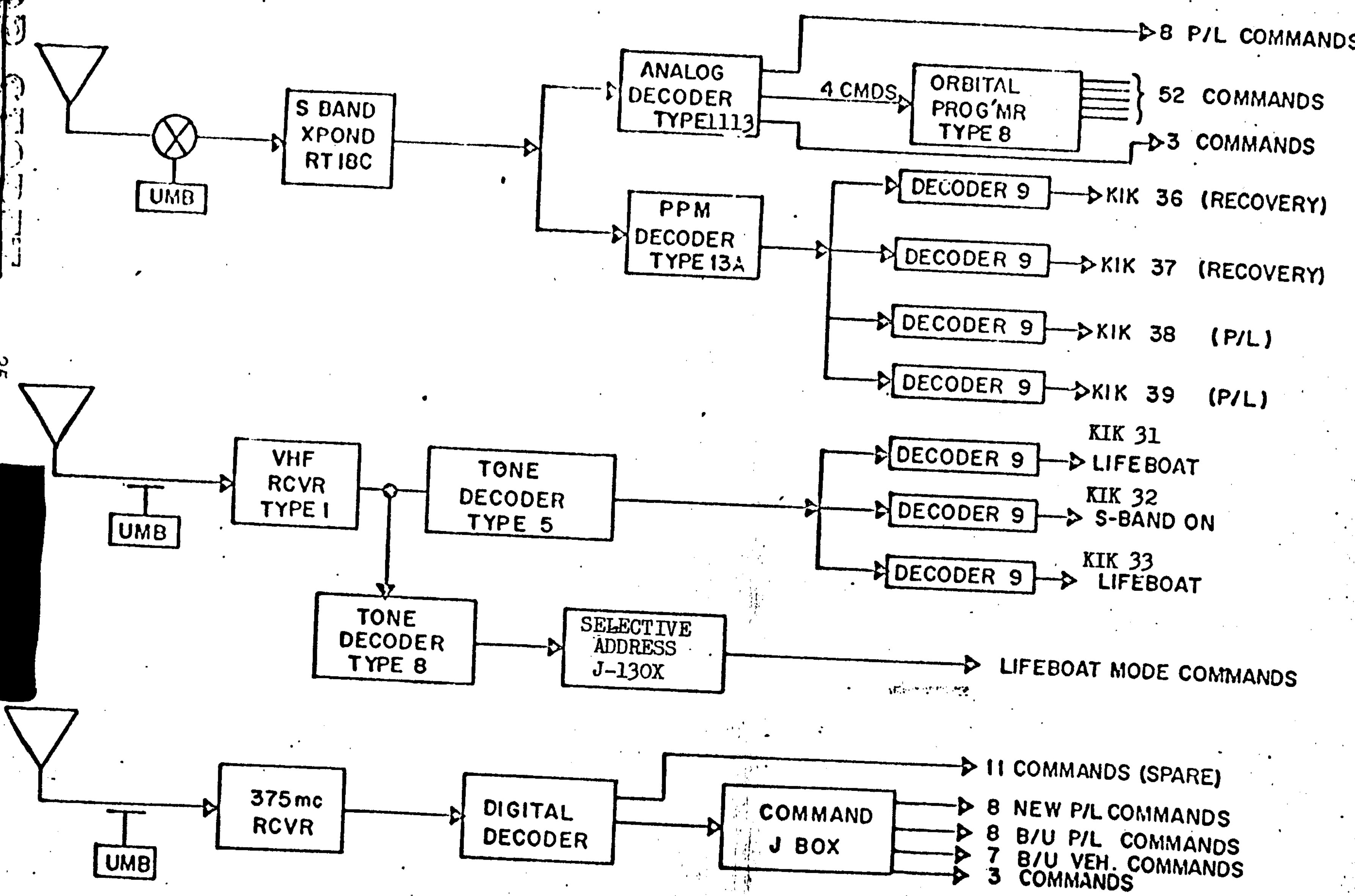
Real-time commands are transmitted to the vehicle by PRELORT, ZEKE, and 375 mhs UHF (UNCLE). The PREcision LOng Range Tracking Radar (PRELORT) operating in the S-Band frequencies interrogates a vehicle-borne transponder for tracking. Commanding is accomplished for the Digital (ZORRO) and Analog Systems by modulation of the S-Band link. The Analog System employs pulse position modulation using combinations of six audio tones. Analog commands are used for selection of programmed payload and vehicle functions, and the ZORRO commands are used to enable the primary reentry sequence and the early "A" to "B" transfer.

The VHF (ZEKE) command system employs a Very-High Frequency (VHF) link where the carrier is amplitude modulated by audio tones. ZEKE commands provide the selection and execution of the Lifeboat back-up recovery sequence. A back-up command capability is supplied by a 375 mhs UHF receiver in conjunction with a 39 command digital decoder. This system

~~TOP SECRET~~ [redacted]

FIGURE 3

COMMAND SYSTEM BLOCK DIAGRAM



~~TOP SECRET~~ [redacted]

supplies back-up commands to the existing analog decoder and supplies at least eight commands to the payload.

A representative list of real-time commands for satellite vehicle function appears in Table IA. Where possible, each real-time command is accompanied by functional telemetry verification in real-time.

Stored commands are provided to initiate vehicle and payload functions during orbital operations and during ejection of recovery vehicles.

An orbital programmer is used to store commands in the vehicle prior to launch. Four reels of punched 35 millimeter, 1.5 mil thick mylar tape provide vehicle and payload functions to be executed at specific times during the mission. Each reel of tape accommodates 13 brushes which make electrical contact with an externally grounded drum through the punched holes, thus providing a capability for 52 programmed commands. Tape speed is nominally 9 inches per subcycle and each tape length can be as long as 192 feet, providing programmed events for approximately 250 subcycles (orbit revolutions). Tape speed and positioning are adjustable by analog commands and back-up ZEKE commands to synchronize the programmer with the vehicle position and orbital period. Accuracy of the orbital programmer is $\pm$ 3.5 seconds including the effect of tolerances on tape punching. A list of stored commands appears in Table IB.

The recovery and Lifeboat timers are solid state components which provide a capability to initiate at least 14 events each. The primary recovery timer is activated by a stored command from the orbital programmer, and

Table I A

Real-Time Commands

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

[redacted]

TABLE IB

J-3 STORED PROGRAM COMMAND LIST

BRUSH	FUNCTION
14	V/h delay reset V/h oblateness start Exposure Control Reset Yaw Programmer Start
17	Dynamic TM Enable. (Instrument operations real-time monitoring)
27	V/h Programmer Start
28	Redundant Off for all Instrument Programs
29	Intermix Position Advance
30-47	Instrument Program ON-OFF, 9 Programs - Even ON, Odd OFF
48	DISIC Independent Mode ON
49	DISIC Independent Mode OFF
50	Exposure Control Start (night to day)
51	Exposure Control Reset (Stop)
52	Exposure Control Start (day to night)

[redacted]

the back-up recovery timer is activated by a ZEKE secure command to control the Lifeboat back-up recovery sequence. A reset function is provided to reset the timer counter to the initial count, and to reset the output relays. Normally, the reset pulse is generated by the timer at the time of its last event. Timer accuracy when installed in a system is plus or minus 0.5 seconds or 0.1 percent between events, whichever is greater.

The satellite vehicle is instrumented to provide timely and accurate data for the pre-launch, launch, orbital and recovery phases of operation. Sensors having the appropriate dynamic range, frequency response characteristics and accuracy are used for data acquisition. Both real-time and stored data is transmitted to the ground station using the Agena telemetry equipment.

Telemetry comprises two separate VHF Frequency Modulated (FM) links. Link I is used primarily to report vehicle and payload status and environmental data. Link 2 is used to telemeter back-up information on payload status, diagnostic data and special payload data. A tape recorder, utilized for the purpose of acquiring data while the vehicle is beyond the range of ground station contacts, is also played back over the ground stations on Link 2. A capability exists for providing two additional telemetry links from the vehicle, if required, by installing the additional transmitters and a four channel multicoupler. For telemetry signal reception and processing, receivers with an IF bandwidth of 300 KC/sec is used. Subcarrier discriminators use IRIG input tuners and standard output low pass filters.

The Agena SS-01B telemetry, as illustrated in Figure 4, uses two separate VHF FM links. Standard IRIG proportional bandwidth FM subcarriers are used for continuous channels and for commutated data. Transmission is in the 215 to 260 MHZ band and conforms to IRIG requirements. Transmitter output power is a minimum of 2 watts (1.5 watts at the antenna terminal).

The maximum subcarrier frequency drift as a result of all causes does not exceed ± 2% of the bandwidth through which the subcarrier's frequency is deviated by full scale data. The subcarrier's frequency deviation is proportional to the modulating voltage (positive frequency deviation for a positive modulating voltage) with a linearity within ± 0.5%. Harmonic distortion does not exceed ± 1%. A design objective is to insure that, under malfunction conditions, no subcarrier oscillator generates an output frequency which interferes with other subcarrier oscillators.

All commutators are compatible with SCF de-commutation equipment. Both non-return-to-zero or return-to-zero pulse train formats are used. Calibration points for at least 0%, 50%, and 100% of the subcarrier bandwidth are provided in each commutator's pulse train. The commutated data is grouped to facilitate the orbital commanding of the vehicle. All commutated data is capable of automatic de-commutation using standard equipment as now provided in the SCF. The total error contribution of any commutator for all combined causes does not exceed ± 1% of full scale.

TOP SECRET

[redacted]

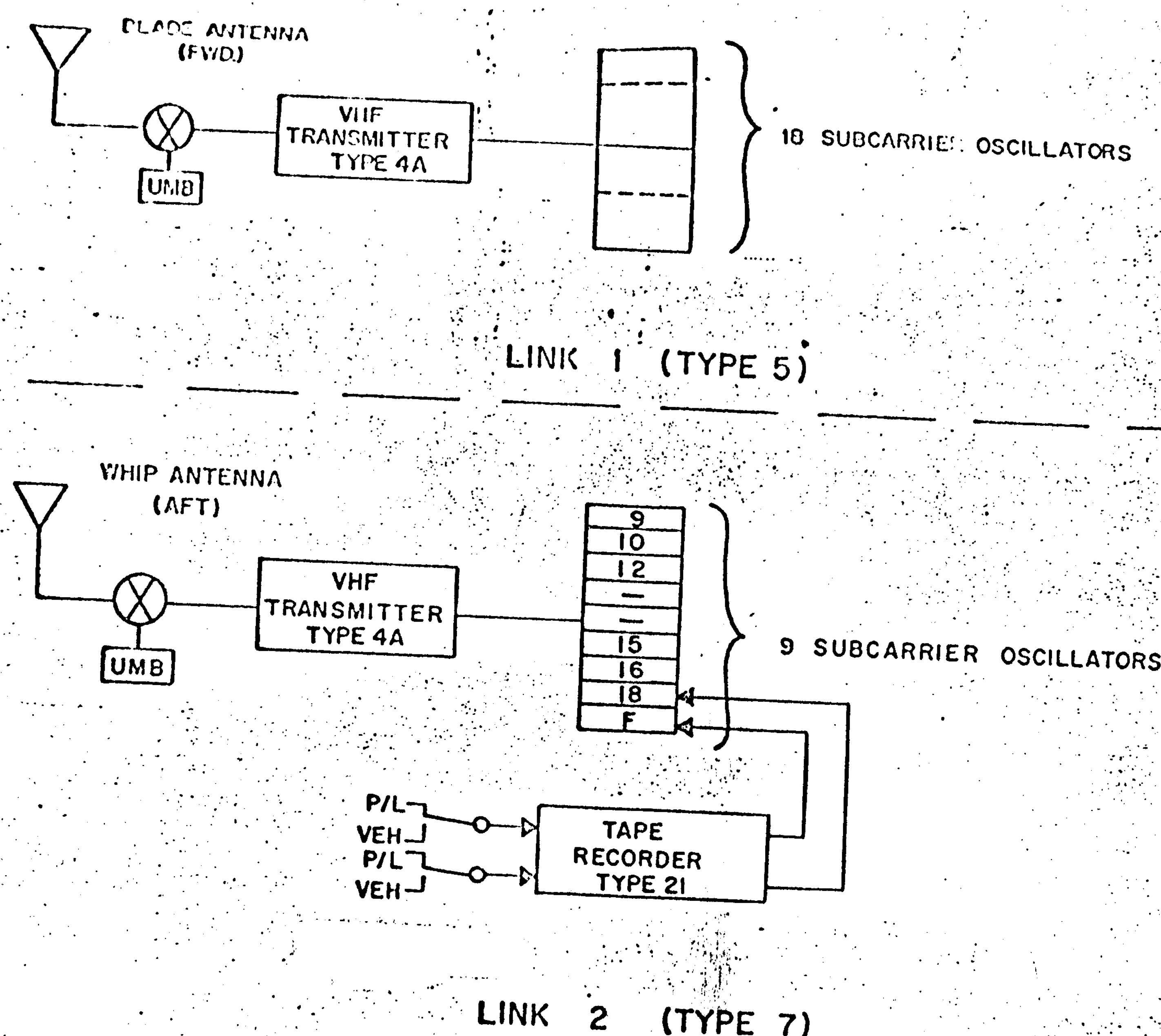


FIGURE 4 TELEMETRY SYSTEM BLOCK DIAGRAM

[redacted]

TOP SECRET

Ten telemetry channels are reserved for the payload. Of these, four are commutated, two on a 0.4 x 60 commutator (24 samples per second), and two on a 5 x 60 commutator (300 samples per second). All critical parameters are instrumented on more than one T/M channel.

A magnetic tape recorder/reproducer, having proven flight reliability, is 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 has dual track data recording capability with a readin-to-readout ratio of 26 to 1. The maximum readin time is 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 is 300 cps, or DC to 60 pps commutated.

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

The primary means of tracking the satellite vehicle is with S-Band radar operated by the SCF. The satellite vehicle contains an S-Band transponder and antenna compatible with the SCF tracking radars. The transponder and antenna system provide a system margin of 3 db. at 875 nautical miles slant range.

Guidance, Attitude Control, and Propulsion

The satellite vehicle is stabilized during on-orbit operation, de-boost to initiate reentry of the satellite recovery vehicles, and orbit adjust maneuvers through the use of the attitude control subsystem. Within the grounds of practicability, the same guidance and control components are used on-orbit as in the de-boost and orbit adjust phases to minimize number of total components and weight.

Throughout the orbital mission, excluding de-boost sequence and orbit adjust maneuvers the satellite vehicle remains 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 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 is subjected to the following momentum unbalances:

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

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

Axis	Torque
Pitch	8 ft. lb.
Yaw	16 ft. lb.
Roll	2 ft. lb.

Pointing accuracy requirements and maximum limit cycle rates are as follows:

Pointing Accuracy and Rates

Function	Requirement	Objective
Pitch Attitude	± 1.5°	± 0.75°
Yaw Attitude	± 2.0°	± 1.10°
Roll Attitude	± 1.5°	± 0.75°
Pitch Rate	.016 deg/sec	.008 deg/sec
Yaw Rate	.016 deg/sec	.008 deg/sec
Roll Rate	.022 deg/sec	.011 deg/sec

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

In addition to the above requirements for stabilization, the satellite vehicle is capable of being maneuvered in yaw. In response to a payload yaw programmer voltage, the vehicle is positioned in yaw to 0.25 degrees per 1.67 millivolt of roll torque input with a stabilization time of 6 minutes.

The de-boost sequence for the satellite vehicle is controlled by signals from a DI/AN recovery timer. Upon the programmed command, the satellite vehicle pitches down a nominal 120 degrees from the local horizontal, while orbiting in the "nose-first" attitude, and holds this attitude with respect to the local horizontal until the recovery vehicle has been ejected. The time required to pitch down is approximately 60 seconds. After recovery vehicle ejection, the satellite vehicle is returned 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 below.

Pointing Accuracy, Pitched-Down Attitude

Function	Requirement (3σ)	Objective (3σ)
Pitch Angle from Local Horizontal	$\pm 7.0°$	$\pm 6.5°$
Yaw Angle from Orbit Plane	$\pm 1.5°$	$\pm 1.1°$
Roll Angle from Radius Vector	$\pm 1.3°$	$\pm 1.0°$

In the event of a malfunction in the primary attitude control subsystem, a back-up stabilization system (Lifeboat) is activated by a secure real-time command. This Lifeboat system is capable of performing all de-boost sequences necessary for properly ejecting one reentry vehicle from the satellite vehicle. Upon initiation, the Lifeboat subsystem is capable of orienting the satellite vehicle from a tumbling mode of 20 degrees per second about any axis, and is 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 is capable of acceptable performance on North to South passes. The ability to perform acceptably on South to North passes is desirable but not presently required. Pointing accuracy required for Lifeboat is $\pm$ 10.5 degrees to the local magnetic vector referenced to the required 120 degree pitch-down orientation. The vehicle is not re-oriented after Lifeboat useage.

An orbit adjust capability to offset atmospheric drag is provided by firing, in boost, one or more of a set of solid rocket motors. The performance characteristics of the orbit adjust subsystem are:

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

Electric Power

The electrical power and distribution subsystem for the satellite vehicle is comprised of a direct current power source, a power distribution network, and power conversion and regulating equipment for both vehicle and payload operation. The direct current power source consists of batteries of sufficient capacity to supply electrical energy to the vehicle and payload from liftoff, through orbital flight, and until separation of the second reentry vehicle.

A capability to provide electrical energy for mission durations of up to 14 days is inherent in the design of the satellite vehicle, but employment is contingent upon mission requirements and the performance available during ascent flight. The number and type of batteries to be carried on any particular flight are based on a detailed electrical loads analysis for that mission. The power source maintains voltage between the limits of 22 and 29.25 volts D.C. measured at the vehicle buss. A power margin of 5 percent based on the predicted -3σ capability of the battery pack for each flight.

The satellite vehicle supplies a maximum of 900 watt hours per day to the payload. This consists of:

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

B. $400 \pm .003$ cycles at $115 \pm 2\%$ volts A.C. with an average load on Phase C of 0.6 amps continuous and peaks of 0.85 amps for periods not to exceed 500 milliseconds (with a duty cycle of 20 minutes on and 70 minutes off); and Phase A with an average load of .1 amps continuous.

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

The electrical wire harnesses provide suitable electrical paths for the distribution of electrical power and signals to satellite vehicle, payload components and major elements.

The maximum voltage drop in any individual circuit from battery to using component (or payload interface), and return, attributable to the harness, including connectors, does not exceed 0.5 volts D.C. Voltage drops in primary leads of up to 1 volt are permissable where this can be shown to be consistent with voltage requirements at the component, and does not involve common wiring resistance of two or more components leading to an interference problem.

The types of batteries which are candidates for each flight power subsystem are summarized with their performance characteristics as follows:

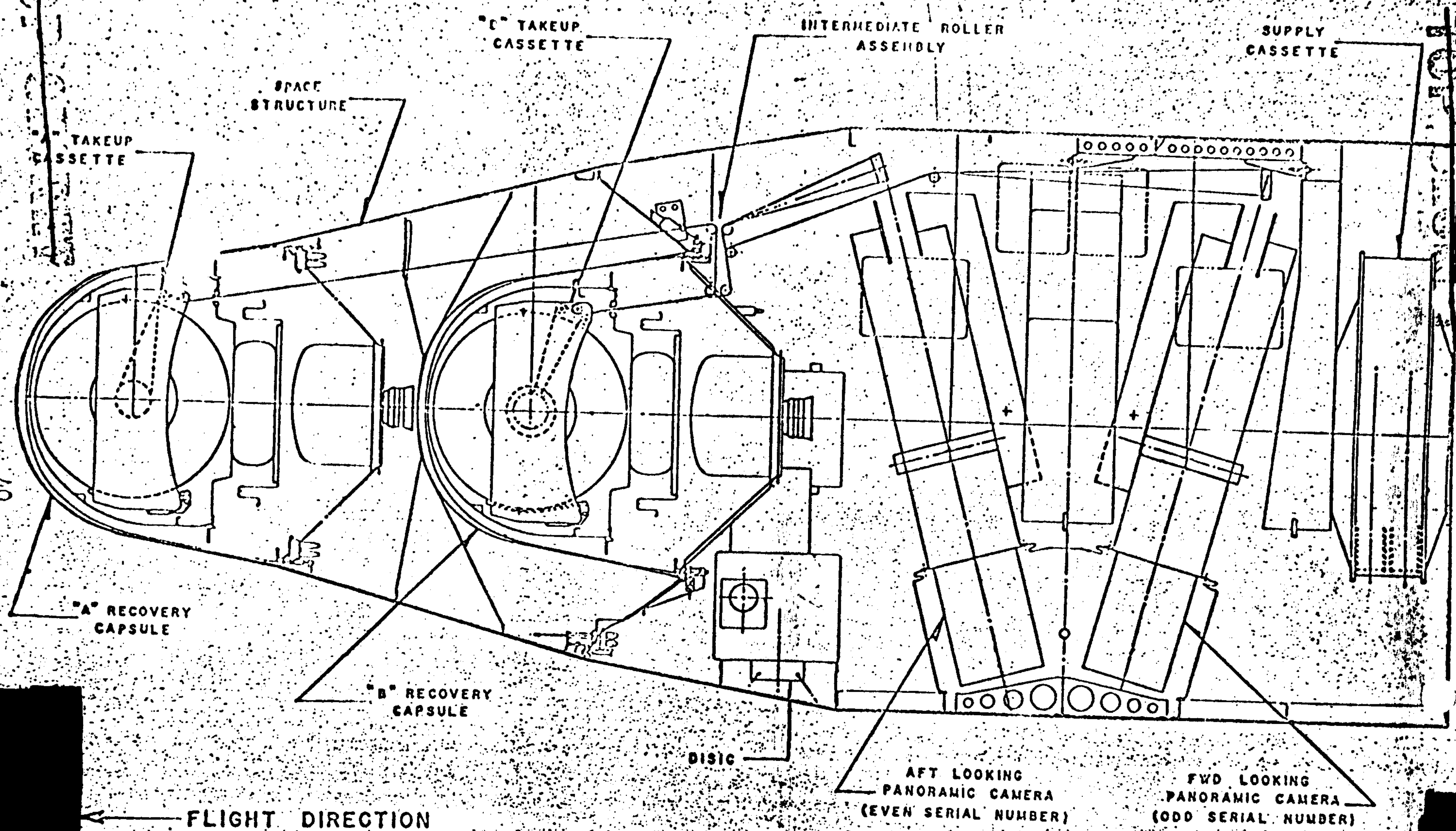
Type	Weight (lbs.)	Energy (-3 sigma) (watt/hours)
VI	28	1,512
1C	119	10,700
1H	126	11,870
1D	107	8,050

All ascent pyrotechnic loads are connected to a separate, diode-isolated primary battery. The isolation doide may be shorted out at the completion of the ascent phase of flight. Fusistors are 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 are capable of handling the maximum all-fire current of the pyrotechnic device. Power and pyrotechnic harnesses may be grouped and routed together, but are separated (as an objective) from harnesses for instrumentation, commands, and test plugs.

The Payload Section

The payload section, as illustrated in Figure 5, comprises a cone-cylinder structure housing the camera equipment, modified Mark 5A Satellite Recovery Vehicles (SRV), and the necessary payload control equipment and electrical cabling. The payload section mates to the forward bulkhead of the satellite vehicle (station 247) as shown in Figure 1. The maximum total payload weight is 1750 lbs. The primary camera equipment consists of two high-acuity panoramic cameras mounted in a 30 degree convergent stereoscopic configuration. Simultaneous operation of both cameras provides stereoscopic photography. A Dual Stellar Index Camera (DISIC) is installed adjacent to the "B" SRV to provide terrain and stellar photography for "A" and "B" missions. Film from the pan cameras is routed through the "B" 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 is cut and take-up, command and T/M functions are transferred to the "B" SRV for the remainder of the mission. Film from the DISIC is routed through an exit housing containing a cut and splice assembly to the "A" SRV. At the completion of the "A" mission, the film is cut and spliced to the leader extending from the "B" SRV, and take-up is initiated for the "B" mission. Transfer of the pan and DISIC take-up control may be accomplished independently by real-time command. Programs for camera on-off operation are provided as stored commands on the orbital programmer. Program selection, camera selection, ~~stereo or mono mode selection,~~ and V/h compensation are provided through real-time commands.

At the conclusion of the "A" mission, the "A" SRV is de-boosted on the desired recovery orbit. Initiation of the "B" mission is not contingent upon prior separation of the "A" SRV. The fairing between the "A" and "B" SRV's is retained on the satellite vehicle until jettisoned by a signal in the de-boost sequence for the second SRV.



[redacted]

J-3 SYSTEM INBOARD PROFILE

FIGURE 5

[redacted]

The payload section provides power distribution networks for all payload equipment forward of the Agena/payload interface. This includes all junction boxes, cables, and connectors necessary for the control and monitoring of payload equipment.

Payload Structural Envelope

The payload equipment is contained within the structural envelope of the payload section of the satellite vehicle. Figure 5 presents the general arrangement of primary payload equipment as well as relative dimensions. The maximum diameter of the payload section does not exceed the outside diameter of the Agena vehicle at the mechanical interface station, and installation of payload equipment external to the payload structural envelope with consequent addition of aerodynamic fairings is avoided. The mounting provisions of the payload structure permit the necessary alignment accuracies required for camera optics and film transport and maintain these alignments under conditions of booster-induced environments during ascent and throughout the subsequent orbital phase. While on-orbit the payload operates in a vacuum environment and under conditions of solar radiation varying from direct sunlight to earth shadow. Environmental control for the operation and survival of the photographic equipment is provided by the payload structural envelope as follows:

A. The cameras and film tracks are provided with a light-proof environment, free of all light leakage that could produce objectionable film fogging.

B. A pressure environment is provided to suppress corona discharge. The Pressure Makeup Unit (PMU) is capable of maintaining pressures of 20 microns or higher in the payload cylindrical sections during camera operation.

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

D. Passive thermal control is provided (where possible) for temperature-critical equipment. The external surface of the payload structure is used for passive temperature control. An optimum absorptivity is provided by surface coatings and mosaic patterns to maintain an average temperature of 70 $\pm$ 30 degrees Fahrenheit inside the cylindrical section of the payload.

Panoramic Cameras

The fundamental purpose of the J-3 Camera Subsystem is to provide extensive stereoscopic photographic coverage of the ground with sufficient detail to allow a photointerpreter to recognize, evaluate, and monitor selected targets. Consequently, the subsystem contains certain features which are designed specifically towards this end. First, the camera uses a high acuity lens in such a way as to take advantage of the high resolution available over a narrow field angle. Secondly, auxiliary horizon recording cameras are mounted in a fixed relationship to the panoramic camera to provide an expeditious means for determining roll and pitch attitude of the panoramic cameras during exposure. A time reference system is provided which allows recovery of the time at which any point in the photographic format was recorded and also the time relationship of horizon optics exposure to panoramic exposure.

The secondary purpose of the camera subsystem is to provide photogrammetric control data having the required geometric accuracy to assist the cartographer in constructing accurate terrain maps from the photography obtained by the system. Of equal importance is the ability to assign accurate geodetic coordinates to the maps so constructed.

For cartographic purposes it is essential to establish the geometric relationship between points on the film format and corresponding ground points. In order to accomplish this, it is necessary to calibrate the internal geometry of the camera. This involves the use of special equipment in pre-flight testing of the system and special data reduction techniques. The calibration information obtained from the tests when performed is supplied to the cartographic community. Additional data are recorded on the film during in-flight photography. These data permit the correlation of the photography with the previously obtained calibration information. Thus, for every point on the film, it is expected that the cartographer can determine two angles, Alpha (across-track or scanning angle), and Beta (along-track angle), ~~with an rms accuracy of 4 arc-seconds each.~~

Camera Description

The complete J-3 Panoramic Camera Subsystem consists of the following:

1. Two identical, 24-inch focal length f/3.5 panoramic cameras, each having two integrated 55 millimeter focal length, f/6.3 horizon optics.
2. One auxiliary structure (supports both panoramic cameras and the electronics packages to form the so-called camera module.)
3. One supply cassette.
4. One supply support structure.
5. Two take-up cassettes.
6. One intermediate roller assembly.

The panoramic cameras are positioned on the auxiliary structure in a V-configuration to provide a 30 degree stereo angle. The auxiliary structure is three-point mounted to the vehicle so that the even serial numbered camera is located forward and views toward the rear (aft-looking), and the odd serial numbered camera is located aft and views forward (forward-looking). The auxiliary structure also provides the mounting surface for the system's electronic packages. The supply cassette, which contains the total film supply for both cameras, is located aft of the camera module. The supply cassette is fastened to its support structure which is, in turn, three-point mounted to the vehicle. Take-up "A", located in recovery vehicle RV-1, and take-up "B", located in RV-2, each take up half of the film of both cameras. The intermediate roller assembly is attached to the vehicle between take-up "B" and the camera module.

The system is basically designed to use 2.5 mil base, 3.0 mil thick, 70 millimeter, EK 3404 film which has an aerial exposure index of 1.6. The supply cassett contains two 28-inch diameter spools, each capable of storing 16,000 ft of film. Each of the two take-up "A" spools is capable of storing 8,000 feet, and each take-up "B" spool is capable of storing 7750 feet of film. The system's total film capacity, therefore, is limited by take-up "B" to 31,500 feet.

The system may be operated with another emulsion type separately or in combination with 3404. These other film types, thickness, approximate aerial exposure index and sensitivity range are listed below.

Type	Thickness (Inches)	Aerial Exposure Index	Sensitivity
SO-380	0.0020	3.6	Panchromatic
SO-121	0.0036	18 (w/W-2E)	Color
SO-340	0.0045	250	Panchromatic
SO-230	0.0030	5	Panchromatic
SO-180	0.0036	18 (w/W-12)	Infra-red
SO-166	0.0045	8000 ASA (no AEI available)	Panchromatic

A summary of the general configuration and operational characteristics are included in Table II.

Camera Operation

The panoramic cameras are independent and similar but are not interchangeable. Each camera consists of its own machined frame upon which most of the camera components are mounted. Because some camera components are attached to the auxiliary structure, the structure must be considered as an integral part of the panoramic camera.

The primary components of the panoramic camera, as shown in the functional schematic in Figure 6, are: (1) drive system, (2) lens, (3) scan head assembly, (4) drum, (5) film transport mechanisms, (6) FMC mechanism, (7) panoramic geometry system, and (8) the horizon optics. The actions

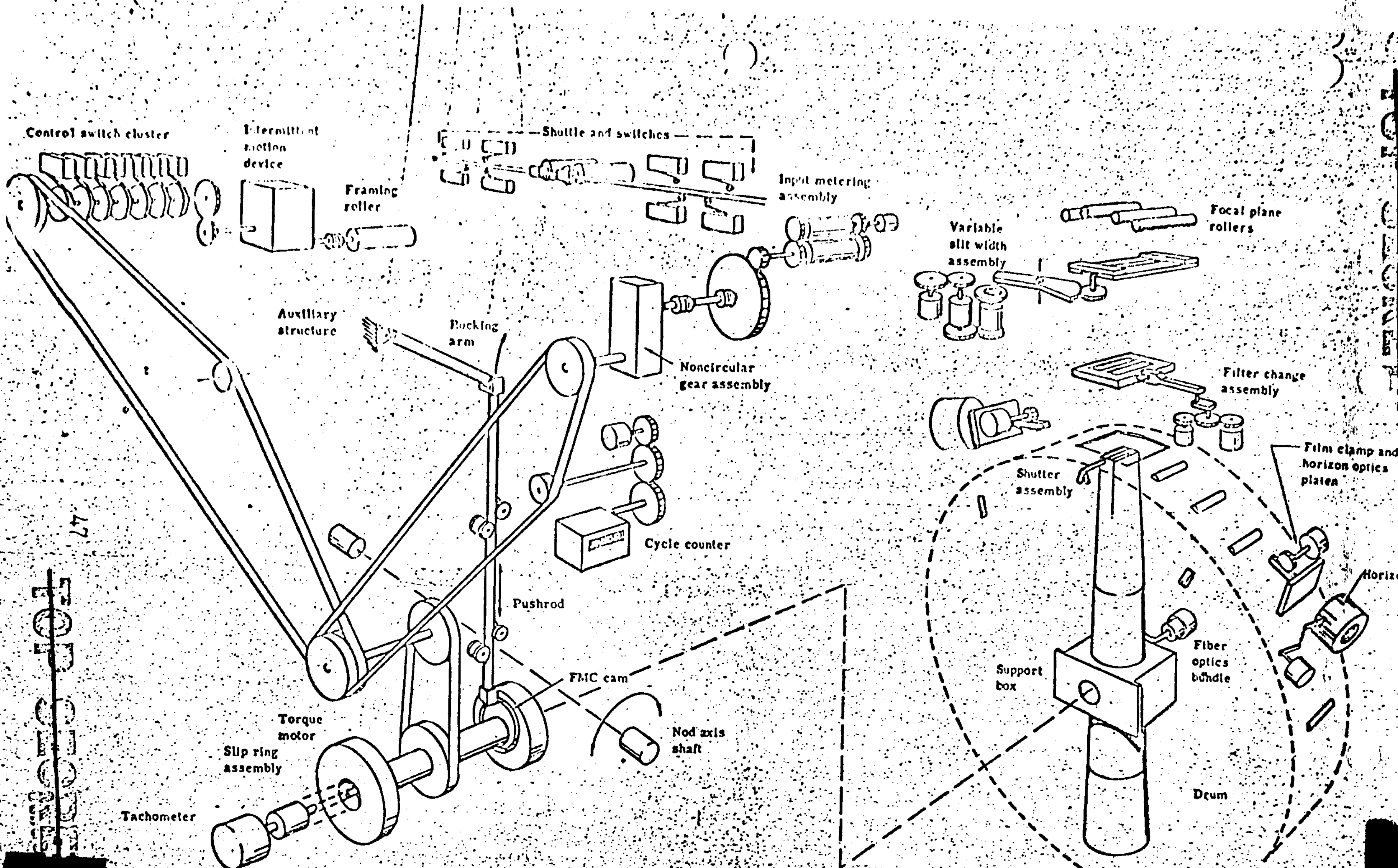
~~TOP SECRET~~ [redacted]

PANORAMIC CAMERA SUBSYSTEM PARAMETERS

PARAMETER	PANORAMIC CAMERAS (2)	HORIZON CAMERAS (4)
Lens	24" f/3.5	55mm f/6.3
T/Number	T/3.85	
Resolution (3404 Emulsion)	110 L/MM C = 0.3	- - -
Usable Format	29.323" x 2.147"	2.1" x 0.9"
Shutter Type	Slit With Capping Shutter	Between the Lens
Angular Coverage	70° x 5.12°	52° x 24°
Nominal Exposure Time	1/400 Second (Variable)	1/100 Second (Fixed)
Filter	Variable - 2 Position Commandable	Wratten 25 Plus Commandable Attenuator
Fiducials	Rail Holes	4 plus one for reference
Cycle Period	1.5 to 4.2 Seconds/Cycle	Every Other Pan Frame
Endlap	7.6%	N/A
Data Recording	Time, Serial Number and Geometery Data	None

TABLE II

~~TOP SECRET~~ [redacted]



CAMERA FUNCTIONAL SCHEMATIC

FIGURE 6

of these components are related and timed through a system of belts and pulleys and special-function gear packages, all of which are driven from a single camera drive motor.

The 24-inch focal length lens is a Petzval design consisting of five elements mounted within a cast magnesium cell. A sixth element, the field flattener, and an exposure/filter device are mounted on the end of a titanium tail cone which is, in turn, secured to the lens cell at the nodal point.

The scan head assembly, which contains the slit width and filter change device and the focal plane rollers, is mounted on the end of the lens cone. This device consists of a bi-directional, four-position slit width changer and a two-position filter changer, and is shown in Figure 7. A slit width failsafe mode or nominal slit width position is also provided. The slit blades are driven by a servo motor which is clutched to a dual potentiometer. A nomograph for determining camera exposure time is shown in Figure 8. The filter and a dual potentiometer are driven by a stepper motor. During the exposure portion of the scan, the focal plane rollers lift the film from the guide rails into the exact focal plane.

In order to prevent light from entering the vehicle compartment through the vehicle/camera interface, a drum rotates with the lens within a network of nonrotating light shields that nod with the drum. The drum itself is light-tight except for the clear aperture end and a smaller opening for the scan head access cover. Two formed pieces of sheet metal, which are attached to the drum around its periphery, rotate inside a labyrinth preventing light from entering alongside the drum. The

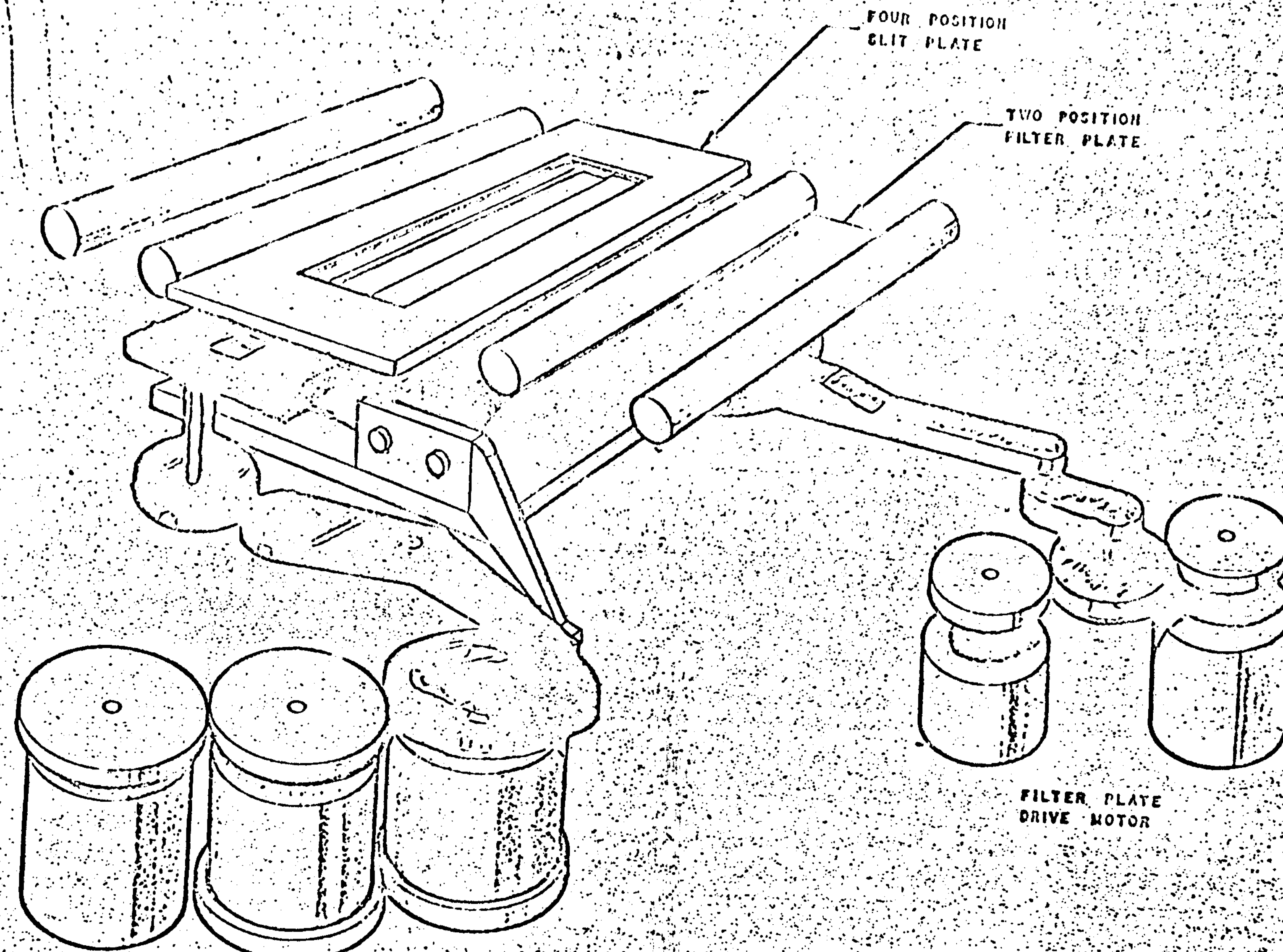

FOUR POSITION
SLIT PLATE
TWO POSITION
FILTER PLATE
FILTER PLATE
DRIVE MOTOR
SLIT OPENING
DRIVE MOTOR

FIGURE 7

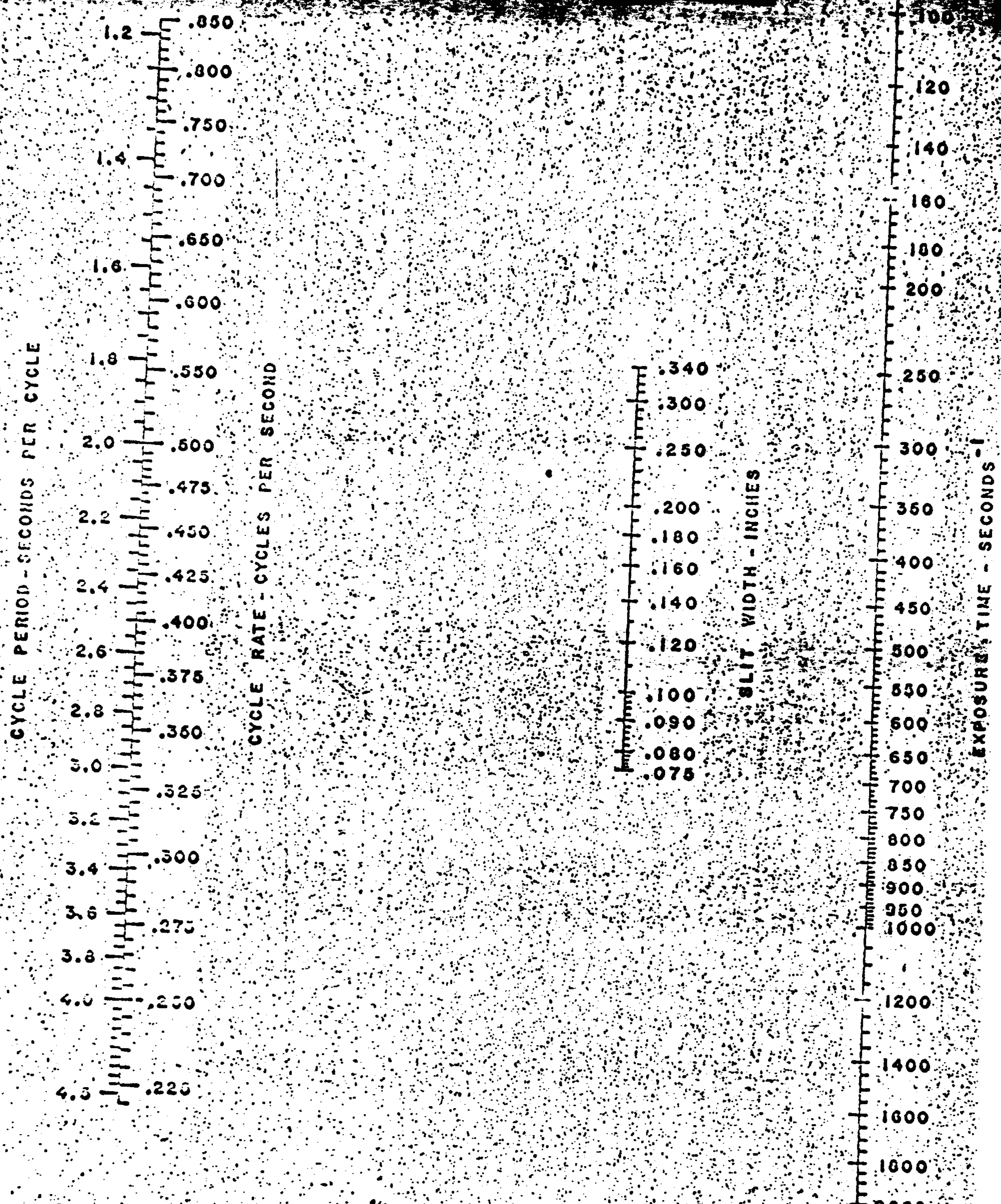


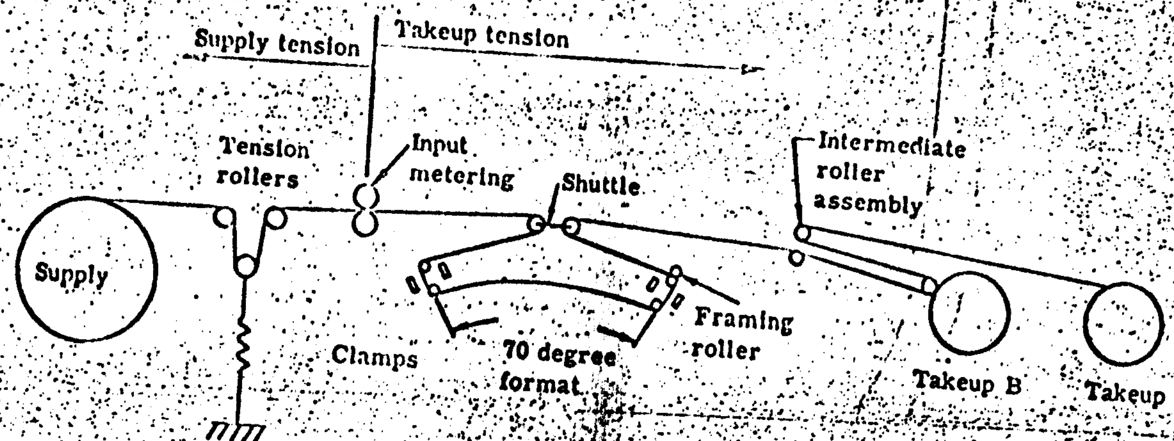
FIGURE 8

SYSTEM EXPOSURE TIME CONVERSION [redacted] inside diameter of the light shields are slightly larger than the diameter of the drum, and the shields encompass the drum over a sufficient portion of the circumference to prevent light from passing around the drum itself.

The drum assembly also serves as a thermal shield for the lens when the camera is inoperative. Furthermore, a series of rollers, located around the circumference of the drum and placed parallel to the lens rotation axis, revolve with the drum just beneath the film guide rails to prevent film from being pulled through the rails. These rollers do not contact the film under normal operation.

The camera film transport system, shown schematically in Figure 9, comprises an input metering roller which is geared through a 99/101 percent clutch to provide continuous input metering at a nominal rate. Film guide rails guide the film over the 70 degree format and film clamps located at either side of the format are actuated during exposure. A frame metering roller pulls one frame of exposed film out of the format area during the non-exposure portion of the cycle. A shuttle mechanism stores the extra length of film arising from continuous film input and output and intermittent frame metering. The shuttle also is used to control the 99/101 percent clutch. The complete film path is shown in Figure 10.

Each camera contains its own FMC mechanism. The FMC mechanism is comprised of a cam, which is driven by the camera drive motor, and a four-bar linkage which is driven by the cam. The linkage is fixed at one point such that the action of the cam against the linkage causes the cameras to



FILM TRANSPORT SCHEMATIC

FIGURE 9

[redacted]

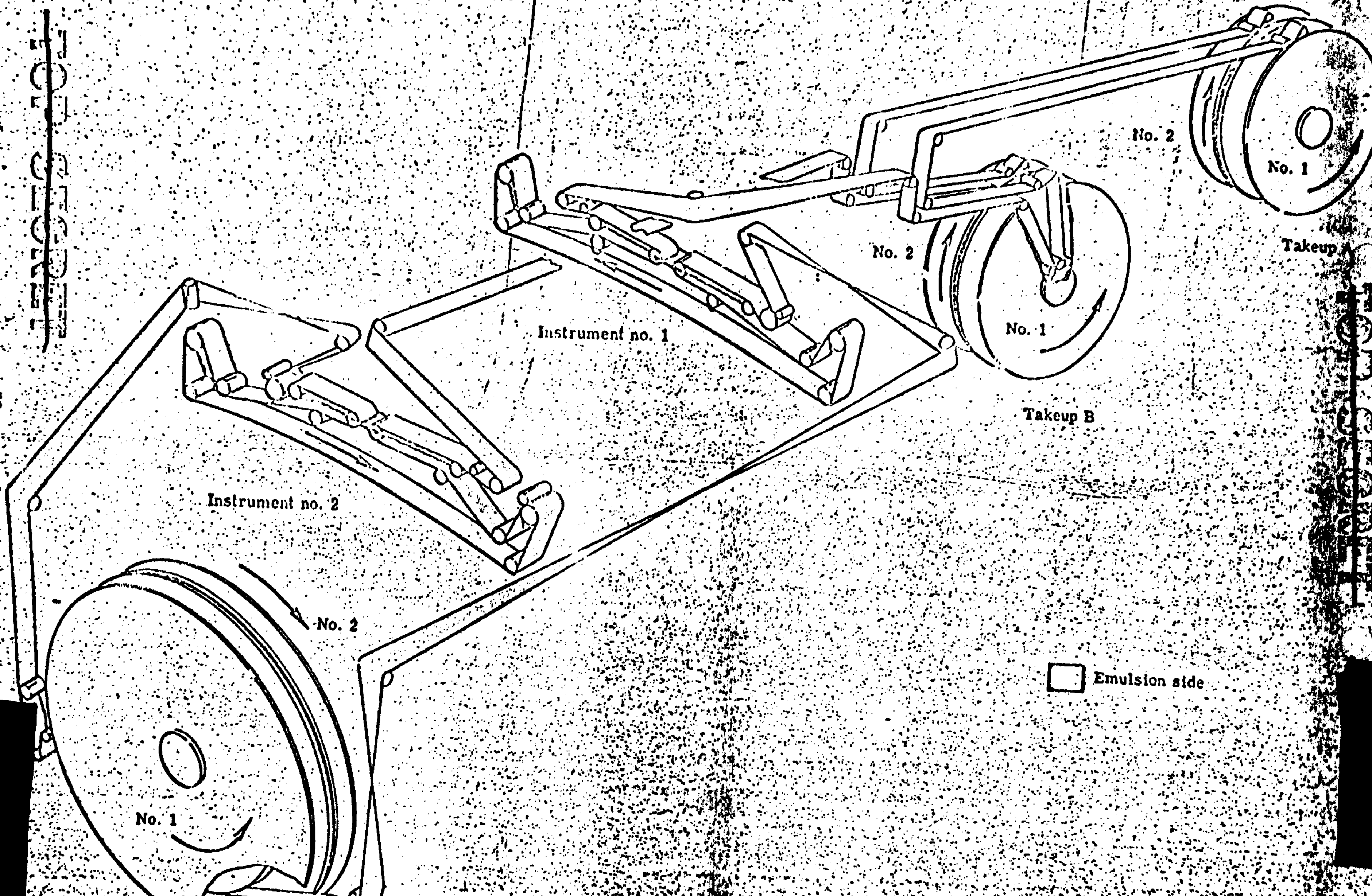


FIGURE 10
FILM THREADING DIAGRAM

[redacted]

rock in the vehicle pitch axis. The FMC cam has been designed to produce a 7.6 percent endlap in each panoramic frame with exact FMC correction.

Each panoramic camera contains two horizon camera assemblies that allow the photogrammetrist to expeditiously determine the pitch and roll attitude of the panoramic camera during exposure. The horizon camera consists of a 55 millimeter f/6.3 lens, a between-the-lens leaf shutter, a shutter-trip solenoid, an attenuator change mechanism, and an assembly housing. The horizon camera assemblies are mounted on each end of the film transport bridge. This facilitates the sharing of a common film supply and path with the panoramic camera. The optical axes of the horizon lenses are nominally, but not exactly, coplaner with the optical axis of the panoramic camera. The horizon cameras operate with every other panoramic camera frame.

The horizon camera uses an integral filter equivalent to a Wratten No. 25. The lens provides a format of 2.1 by 0.9 inches. The corresponding half angles are 26 and 12 degrees, respectively.

The horizon camera housing provides a support structure for the lens, shutter mechanism, lens cone, lens hood, and filter change mechanism. The filter change mechanism, mounted in front of the lens, consists of a sliding filter on a track, a drive motor, and connecting linkage. An attenuating filter may be slid in front of the lens when films faster than the basic 3404 are used.

The relationship of the exposure functions within the camera are controlled by series of cams. The one revolution per cycle cam operates the panoramic camera functions while the one-half revolution per cycle cam controls the horizon cameras. The sequence of events is shown in Figure 11.

Camera Format

A sketch of the camera format with associated data is shown in Figure 12 and the SLP data block format is shown in Figure 13. The relationship of the camera formats to the ground scene is shown in Figure 14. The useable pictorial area of the panoramic camera photography is 29.323 inches along the film major axis and, as shown, 2.147 inches across the film. The coverage associated with this format size is listed in Table III for various altitudes.

The photographic scale can be approximated by using the nomograph in Figure 15. This chart includes the scale variation with departure from the center of the format. The center of format is denoted by a dual rail hole as shown in enlargements "A" and "B" on Figure 12.

The horizon camera format includes five fiducials which register as 0.006 inch diameter dots. These dots are formed by 1.5 volt incandescent lamps which contact print through small holes in the format frame when the panoramic camera is in the center of format. The fiducials are nominally positioned to provide the basis for a coordinate reference system for calibration. The intersection of imaginary lines joining opposite fiducials represents the origin of the coordinate system. (This intersection does not represent a photogrammetric point.)

PANORAMIC CAMERA SUBSYSTEM COVERAGE
(3.0 MIL BASE FILM)

Altitude (N.M.)	80	85	90	95	100	105	110	115	120
Frame Forward Cover (N.M.)	7.7	8.2	8.6	9.1	9.6	10.1	10.6	11.0	11.5
Frame Width Cover (N.M.)	116.0	123.2	130.5	137.7	145.0	152.2	159.5	166.7	174.0
Area Per Frame (Square N.M. x 10^2)	8.9	10.0	11.3	12.5	13.9	15.3	16.8	18.4	20.0
Mission Stereo Cover (Square N.M. x 10^6)	4.9	5.6	6.2	7.0	7.7	8.5	9.3	10.2	11.1

100
10
1000

2.5 640 150 100

640
1600
6 40000

TABLE III

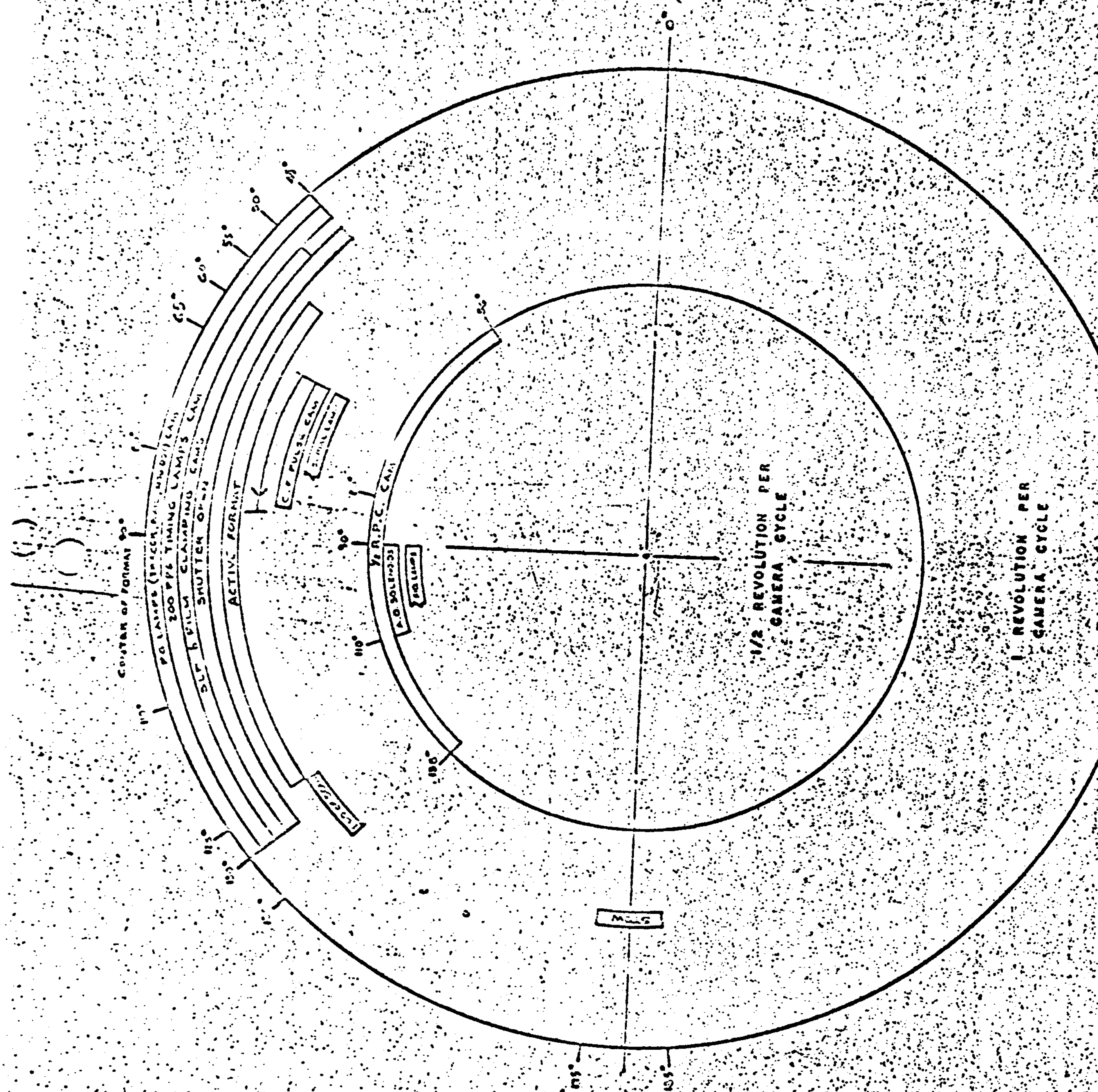
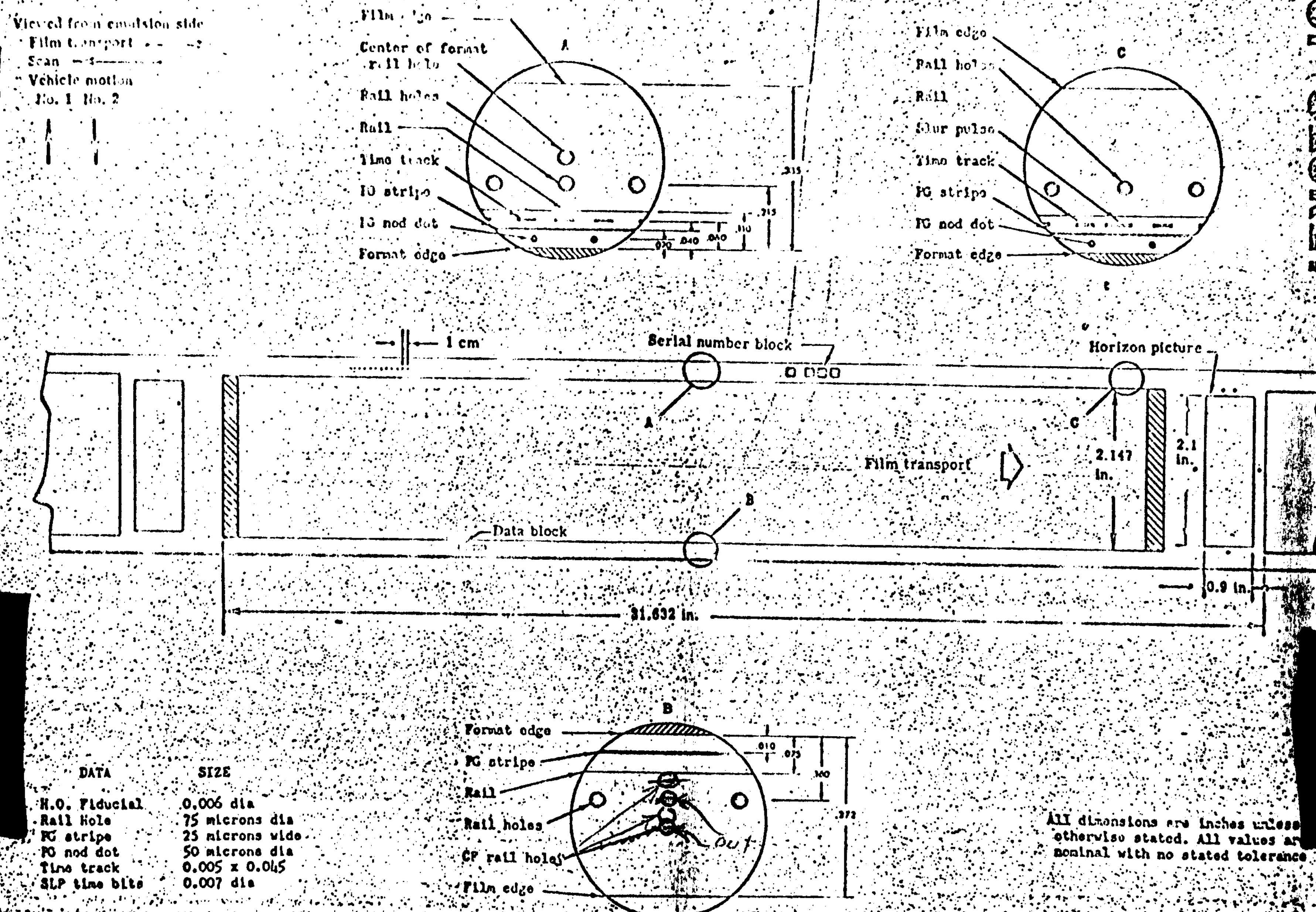


FIGURE 11
TIMING DIAGRAM

~~TOP SECRET~~

[redacted]



DATA	SIZE
H.O. Fiducial	0.006 dia
Rail Hole	75 microns dia
PG stripe	25 microns wide
PG nod dot	50 microns dia
Time track	0.005 x 0.045
SLP time bits	0.007 dia

All dimensions are inches unless otherwise stated. All values ar[e] nominal with no stated tolerance

[redacted]

FIGURE 12

ORIGINAL NEGATIVE FORMAT DRAWING

~~TOP SECRET~~

[redacted]

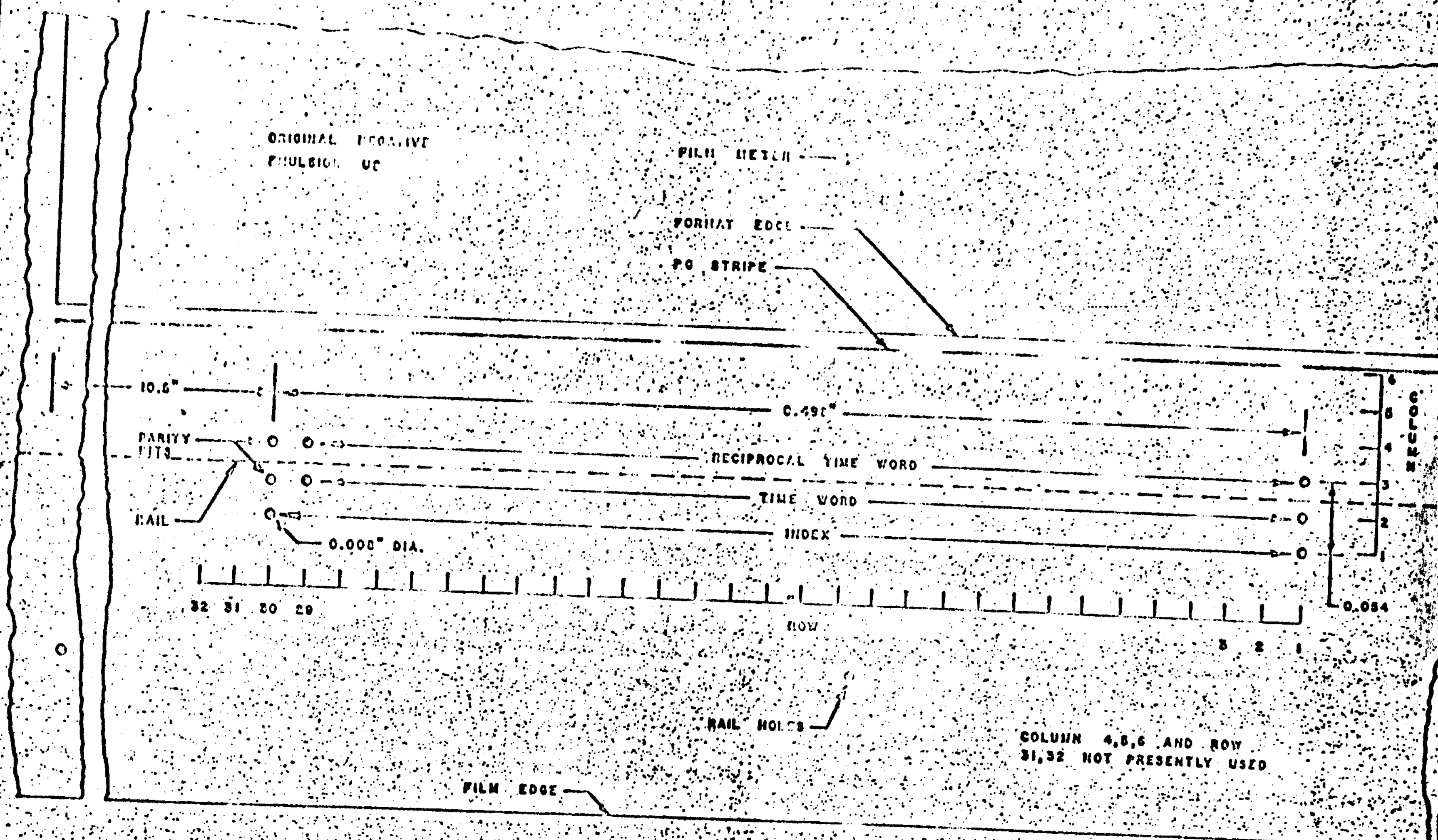


FIGURE 13
DATA BLOCK FORMAT

[redacted]

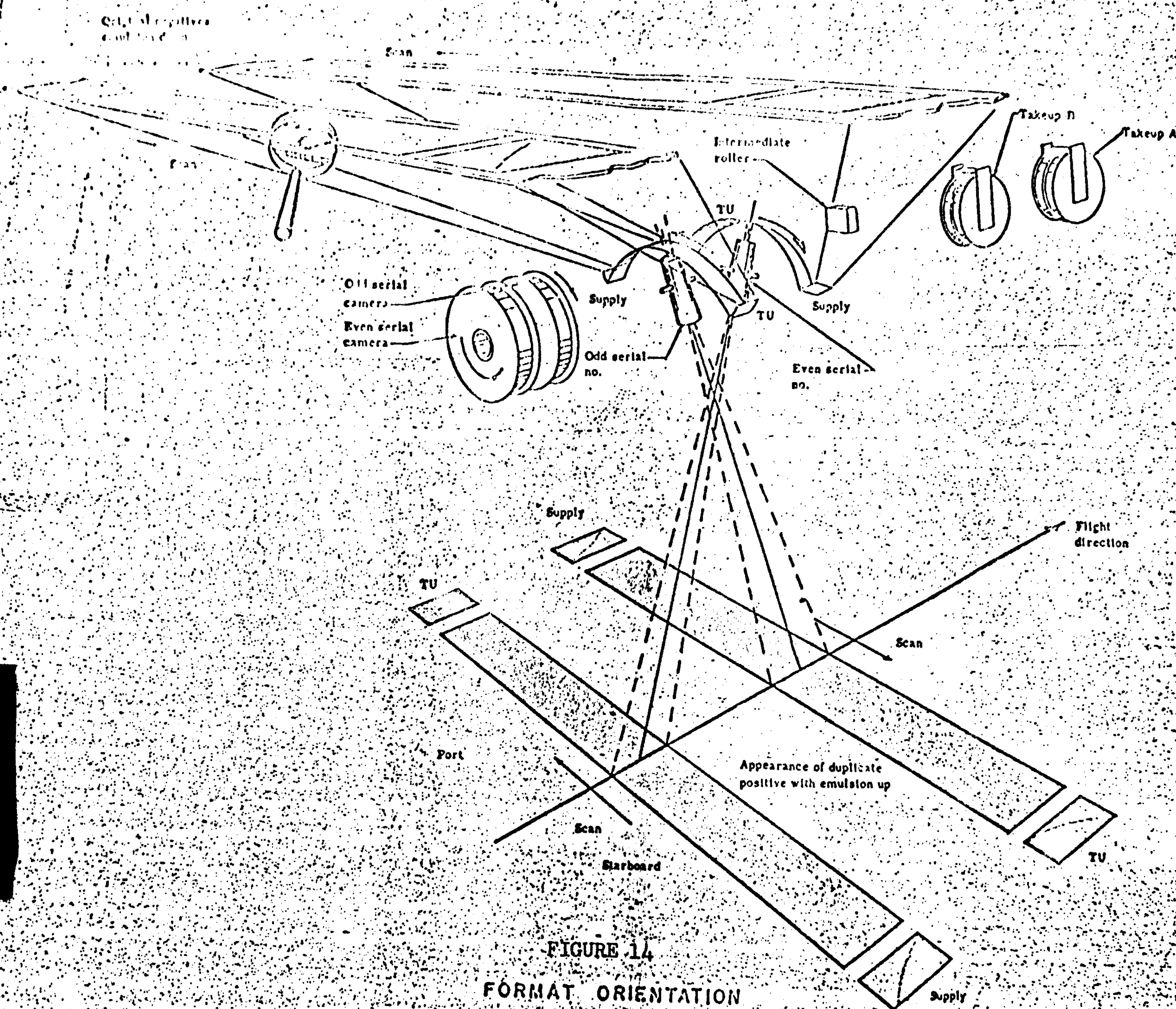


FIGURE 14
FORMAT ORIENTATION

[redacted]

SCALE CONVERSION

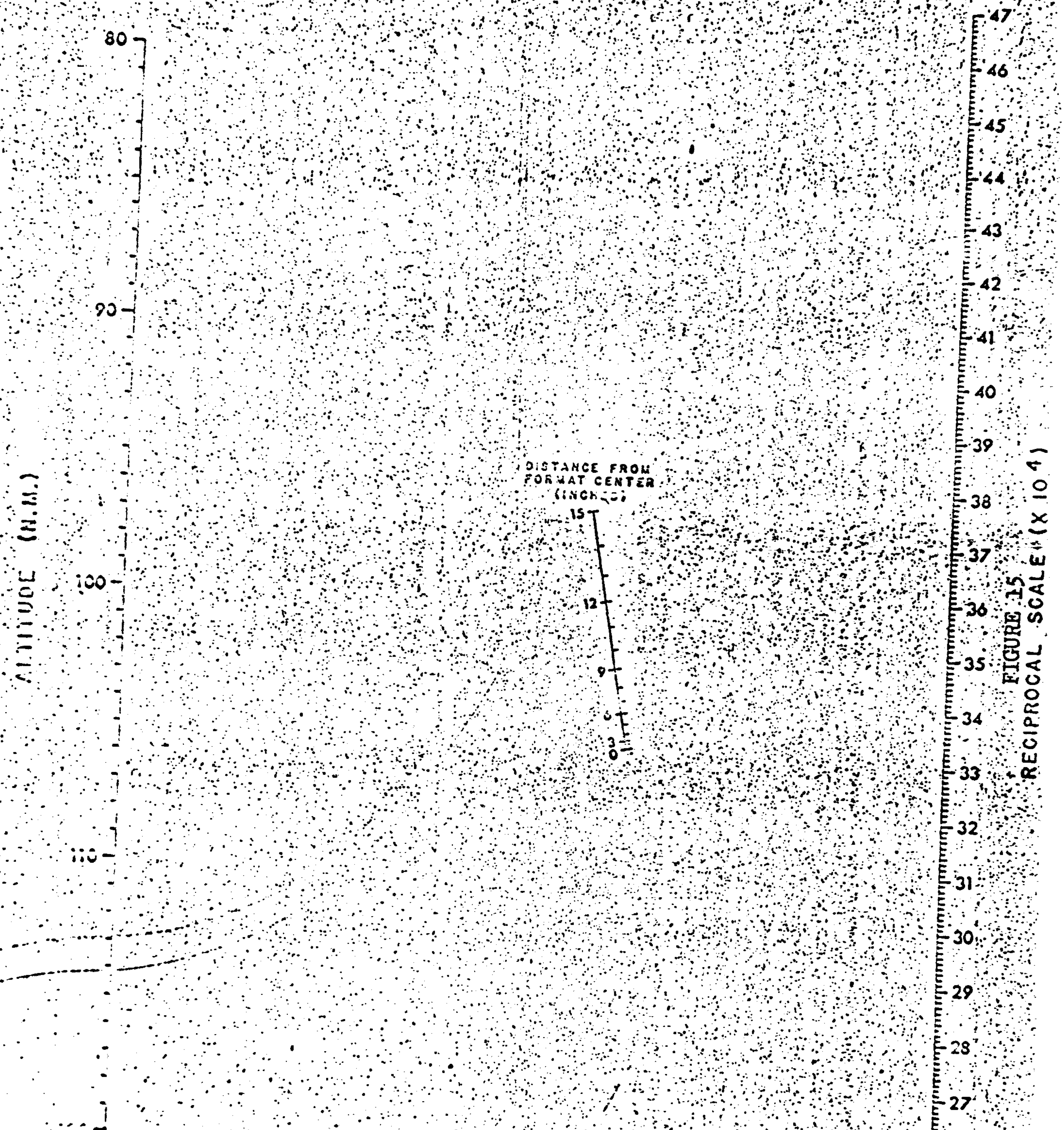


FIGURE 15

[redacted]

Panoramic Geometry

The camera incorporates the means by which a calibration framework can be developed that allows the panoramic photography to be utilized for cartographic purposes. The components of this panoramic geometry framework internal to the camera are: (1) a series of holes spaced at one centimeter intervals along each film guide rail, (2) a pair of lamps fastened rigidly to the upper end of the lens scan arm which produce traces on either side of the format and which represent the locus of the path of the principle axis of the lens, (3) a nod angle to scan angle calibration system which, by means of a xenon flash triggered by an optical encoder mounted on the nod axis, images a series of small dots along the edge of the format, and (4) a series of timing pulses also imaged on the edge of the format to provide a time reference. The location of this data on the film is shown in Figure 10.

To obtain the initial calibration of each camera, a grid which has been very accurately scribed onto a thick glass platen is exposed onto a length of film. This film is then programmed through the camera and the above noted data is double exposed onto the grid. An accounting is made for the distortions due to the printing of the calibration grid, dynamics within the camera system, and developing processes. After these are factored out, it is possible to calibrate the rail holes through the grid intersections in such a way as to allow the use of the rail holes as the basic framework actual operation for the determination of the Alpha (scan) angle (since the grid is not a permanent part of the camera).

When their position is related to the xenon flash nod-to-scan calibration dots, the relationships of nod-to-scan angle can be determined. The Beta angle can best be determined by a direct measurement from the PG stripes which define the principal point of the lens at any instant in time.

To make proper use of the material for cartographic purposes, it is also necessary to know the attitude of the vehicle during exposure. This information is provided by the DISIC, as described in Section IV, which simultaneously records stellar and terrestrial imagery and therefore allows for a determination of the vehicle attitude at that instant. In addition to the internal geometry for each camera, it is necessary to know the relative orientation of one panoramic camera to the other, as mounted in the vehicle. This will be established after vehicle installation by means of an external array of calibration collimators.

The panoramic geometry components record a sufficient amount of accurate data on each panoramic frame to enable the cartographic community to determine, at every point on the frame, the following parameters:

1. Alpha (scanning) and Beta (along-track) angles with an accuracy of 4 arc seconds, 1 sigma (rms).
2. The absolute time of exposure with an accuracy of 2.5 milliseconds, 3 sigma. (This is necessary since all the images in the panoramic format are not photographed simultaneously.)
3. The time of exposure relative to another specific point on the format with an accuracy of 1 millisecond, 3 sigma.

4. The change in the nod angle of the camera, at the time of exposure of the image point, from a pre-determined nod angle, ~~with an accuracy of 4 arc seconds, 1 sigma~~.

These parameters can be obtained for each image point by making linear measurements between the image point and the data which is recorded at the edge of the film adjacent to the format. A row of rail hole images and solid traces are recorded along the length of the format (scanning direction) at one edge of the film. At the other edge of the film, a row of rail hole images, a solid trace, a row of timing marks, and a row of nod angle dots are recorded.

The rail hole images are round spots about 75 microns in diameter. They are the images of 0.0015 inch diameter holes in the rails. The rails support the edges of the film on an arc of 70 degrees. There are 73 holes in each rail spaced about 1 degree apart angularly and approximately 1 centimeter apart linearly.

The rail holes are calibrated metrically so that their position is known to about 6 microns, 1 sigma (4 microns in each of two orthogonal directions), so that this information, when used in conjunction with rail diameter measurements, allows determination of the Alpha angle (scan angle) of each hole. The accuracies of the rail hole calibration vary slightly between cameras, but are specified in the calibration report provided with each camera.

In each rail, the rail holes are exposed one at a time as the lens rotates about its nodal point. The rail lamps are positioned in such a way that the rail holes and the ground images are exposed simultaneously. The zero scan angle is set arbitrarily by an additional rail hole at the center of each of the rails.

The solid traces, usually referred to as the panoramic geometry traces, are useful in locating the principal point of the lens. The position of the principal point must be known so that linear measurements in the x-direction of the film (along-track) can be correctly converted to Beta angles. The distances of the two light spots which produce the solid traces from the principal point will be measured and given as x, y, coordinates. The principal point will have (0.0) coordinates. The panoramic geometry traces are about 25 microns wide. The x, y, coordinates of the two light spots that produce the solid traces will be given with respect to the principal axis from measurements made on an optical bench.

The nod dots are round spots about 50 microns in diameter. They are produced by a separate subsystem consisting of an accurate optical encoder, electronic circuits, an xenon flashtube, two sections of optical fiber bundles, a rotating optical coupling, and a lens.

An optical encoder is mounted on the nod shaft of the designated camera to measure the nod angle. At certain fixed nod angle positions (every 19.78 arc seconds), the encoder and its associated electronics generate electrical pulses. These pulses are introduced ~~to a count-down circuit one pulse every 39.55 arc-seconds of nod shaft rotation.~~ These ~~remaining~~ pulses trigger the xenon flashtube which flashes for 2 microseconds in response to each pulse. The light of the xenon flashtube is piped by the fiber optics bundles through the rotary optical coupling to the scan head, where the fiber bundle is masked except for a very small hole. The light emerges from this hole and is projected by a small lens onto the film.

In this manner, dots which can be identified with definite nod angles are recorded on the film. The position of a nod dot along the film (Y coordinate of a frame) depends on the location of the scan head with respect to the rails (in other words, the scan angle) when the xenon flashtube was triggered. This provides sufficient information to obtain the calibration of the nod versus the scan angle. The zero nod position is arbitrary, and it is defined by an additional nod dot generated by the zero reference output of the encoder.

The time marks are elongated spots 0.005 inch wide by 0.045 inch long. Their main purpose, as part of the scanning function, is to facilitate the determination of the time of exposure of the format. The time marks are generated by an accurate pulse generator (200 pulses per second) which triggers a neon tube. The light of the neon tube is focused on the film by the same lens which focuses the nod dots. The time marks permit the determination of the time difference between the exposures of two different points on the format with an accuracy of 1 millisecond (3 sigma).

Data Displays

In addition to the panoramic geometry display, certain other data appears on the film. The vehicle clock is read out to a silicon light pulser block (SLP) which exposes the time on the film in binary form. The binary spot size is about 0.007 inch in diameter. There are actually six columns of 32 bits available, but only three columns of 30 binary bits are used as shown in Figure 11. The columns are parallel to the edge of the film and are read from left to right as seen from the side of the film away from the emulsion when the SLP is on the edge of the film nearest the viewer. The column furtherest from the film edge is column number one, and all 30 bits are illuminated to provide a registration for mechanical readout. Column two presents the time work in rows 1 through 29 with the 30th bit being the parity bit. Column three presents reciprocated time, again with the 30th bit being the parity bit.

To properly image the SLP, it is necessary to firmly clamp the film to the block during exposure. Since this is not possible in the film format area, the SLP block is located on the take-up side of the framing roller. This means that any time readout as seen on the film is associated with the following (next higher number) frame, or conversely, when ascertaining the time a particular frame was taken, it is necessary to look at the SLP readout block on the previous or lower numbered frame.

Determination of the instant during scan when the time was recorded is made by noting the ~~position~~ leading edge of a smear pulse in the 200-cycle-per-second timing marks. This is a variance on the J-1 technique where clock interrogate was referenced by a blanked pulse.

The camera serial number is exposed on the film margin opposite to the data block as shown in Figure 12. The serial number is located approximately 6.9 inches to the left of format center. A cross is exposed in this same margin at the initiation of camera operations. This cross is physically located adjacent to the camera serial number. ~~The exposed image on the film can be located at any point along the film edge as in the case with J-1 systems.~~

The margin of the horizon camera format contains five fiducial hole images which are used to reference the position of the principal point of auto-collimation of the horizon lens.

Camera Calibration

The panoramic camera lenses and horizon cameras are individually calibrated prior to being mounted on the panoramic camera. This individual calibration consists of determining the principal point of auto-collimation and the equivalent focal length, and checking the lens distortion characteristics. Subsequent to this, each camera system is calibrated to determine the position of the horizontal cameras in relation to their respective panoramic camera lens. The accuracy of these calibrations is shown in Table II.

DISIC

The J-3 System contains a cartographic frame camera which records terrain

PANORAMIC CAMERA CALIBRATION

COMPONENT	PARAMETER	CALIBRATION
Main Lens	Equivalent Focal Length	25 Microns
Main Lens	Radial Distortion	1 Micron
Main Lenses	Convergence	60 Arc Seconds
Horizon Optics	Equivalent Focal Length	25 Microns
Horizon Optics	Principle Point to Fiducial Intersection	10 Microns
All Lenses	Alignment	60 Arc Seconds

TABLE IV

and stellar photography. The DISIC (Dual Stellar Index Camera) is comprised of three cameras: one downward looking terrain camera and two stellar cameras whose axes are 10° above the horizontal. The DISIC inboard profile and end view installation drawing are shown in Figures 16 and 17, respectively. A listing of the general camera design parameters is given in Table V.

The DISIC subsystem is designed to operate in conjunction with the panoramic cameras, termed the slave mode, or to operate independently. The camera functions are controlled within the unit and are time phased rather than controlled from signals from the panoramic camera. There are therefore no indications on the panoramic film to show DISIC operations. All frame correlation is accomplished with the interpolation of the binary time words on the panoramic, terrain and stellar films. The mechanical block diagram of the DISIC is illustrated in Figure 18.

DISIC Operation

15, 625, or 18.75

The terrain camera is preset to operate at either 9.375, ~~or~~ 12.5 seconds per cycle based on planned camera altitude. The stellar camera cycle period is 3.125 seconds during slave operations (Mode I) thus producing three stellar frames for each terrain frame at the 9.375 second period. When the DISIC is in the independent mode (Mode 2) the stellar camera operates once for each terrain exposure. The sequence of camera events is shown in Figure 19 for a typical 9.375 second terrain camera cycle period.

An exploded view of the functional schematic and film threading schematic of the camera is shown in Figure 20. All of the motions within the camera are produced by a single drive motor. Certain options have been incorporated into the design to increase operational flexibility and

[redacted]

DISIC SYSTEM PARAMETERS

PARAMETER	TERRAIN CAMERA	STELLAR CAMERA (2)
Lens	3-Inch Ikogon	3-Inch Ikotar
Aperture	f/6.3 f/4.5 , T/7.3 T/6.75	f/2.8 T/3.1
Static Lens Resolution (3400 Film)	60 L/MM AWAR, C = 0.3 Wratten 12 Filter	[illegible] L/MM AWAR, C = 3.0 (3401 Film)
Film Format	4.5 Inches by 4.5 Inches	1.25 Dia. with Flats
Angular Coverage	74° by 74°	23-1/2°
Lens Distortion (Max.)	30 Microns (R) 5 Microns (T)	15 Microns (R) 5 Microns (T)
Film Flattening	By Glass Plate	By Glass Plate
Reseaux	2.5mm Spacing 10 Microns Max. Width on Glass plate	2.5mm spacing 10 Microns Max. Width on Glass plate
Natural Fiducials	1 Set of Four	1 Set of Four
Shutter Type	Rotary	Rotary
Selective Exposure Time	1/250 Second 1/500 Second	1.5 Seconds (Fixed)
Cycle Period	9.375 Seconds/Cycle 12.5 Seconds/Cycle 15.625 18.75 - - -	3.125 Seconds/Cycle (Mode 1) Same as Terrain (Mode 2)
Dual Stellar Operation		Simultaneous or by Selection
Knee Angle		100°
Data Recording	Time & Camera Serial Number	Time & Lens Serial Number
Film Requirements	5-Inch by 2,000 Feet	35mm by 2,000 Feet
Film Weight (2.5 Mil. Base)	18.3	5.3 Pounds

TABLE V

[redacted]

~~TOP SECRET~~

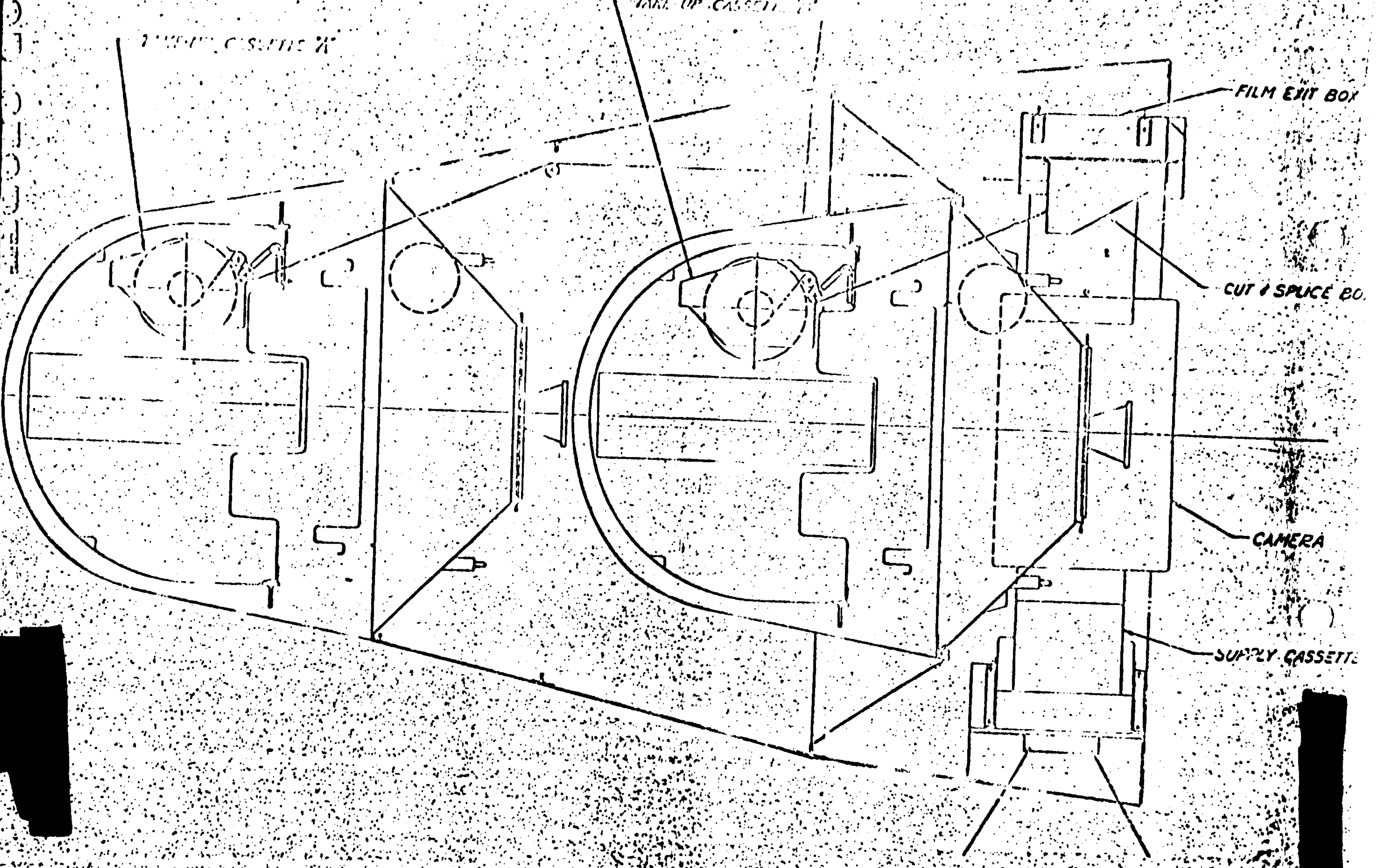


FIGURE 16

~~TOP SECRET~~

[redacted]

[redacted]

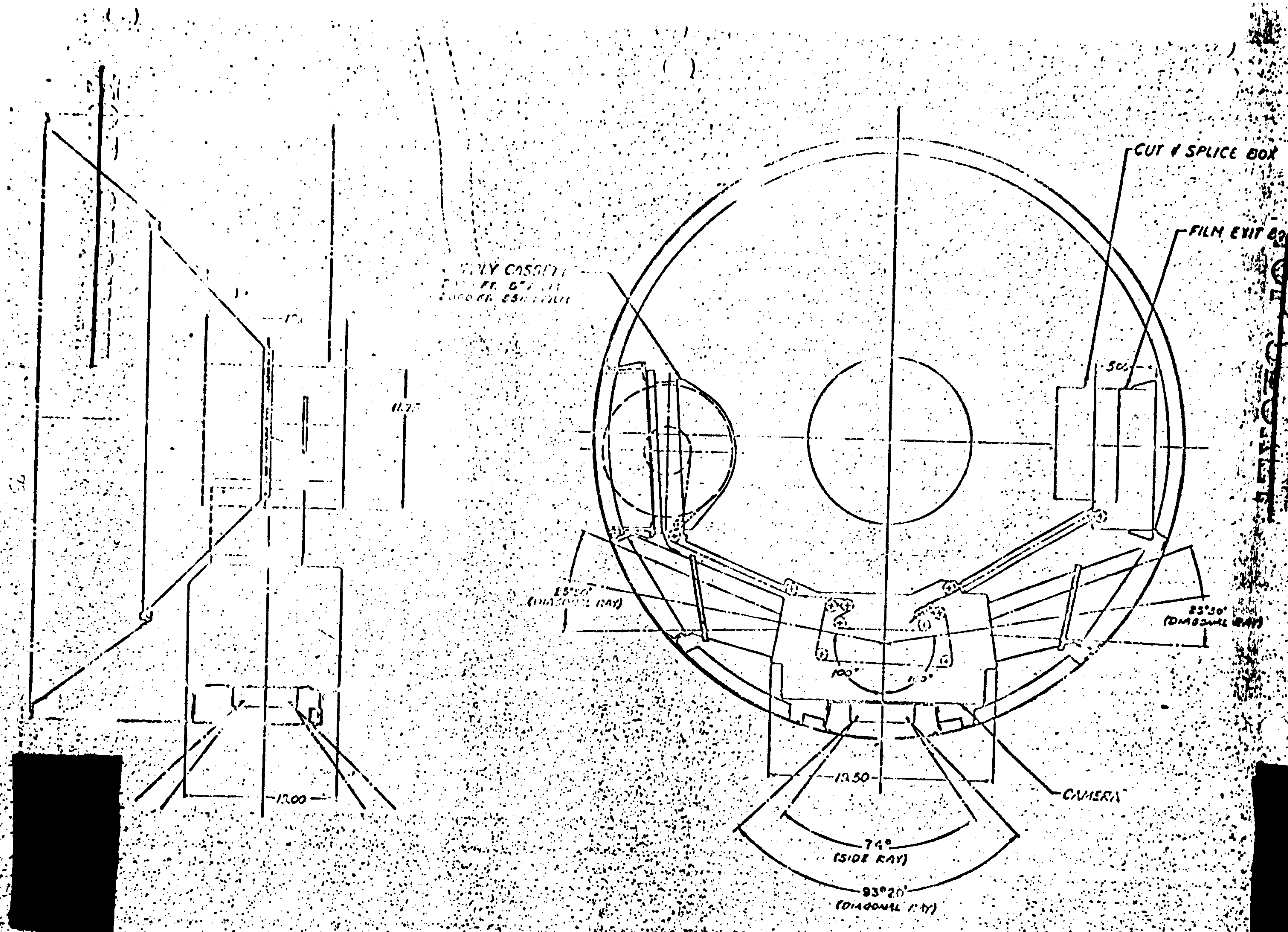
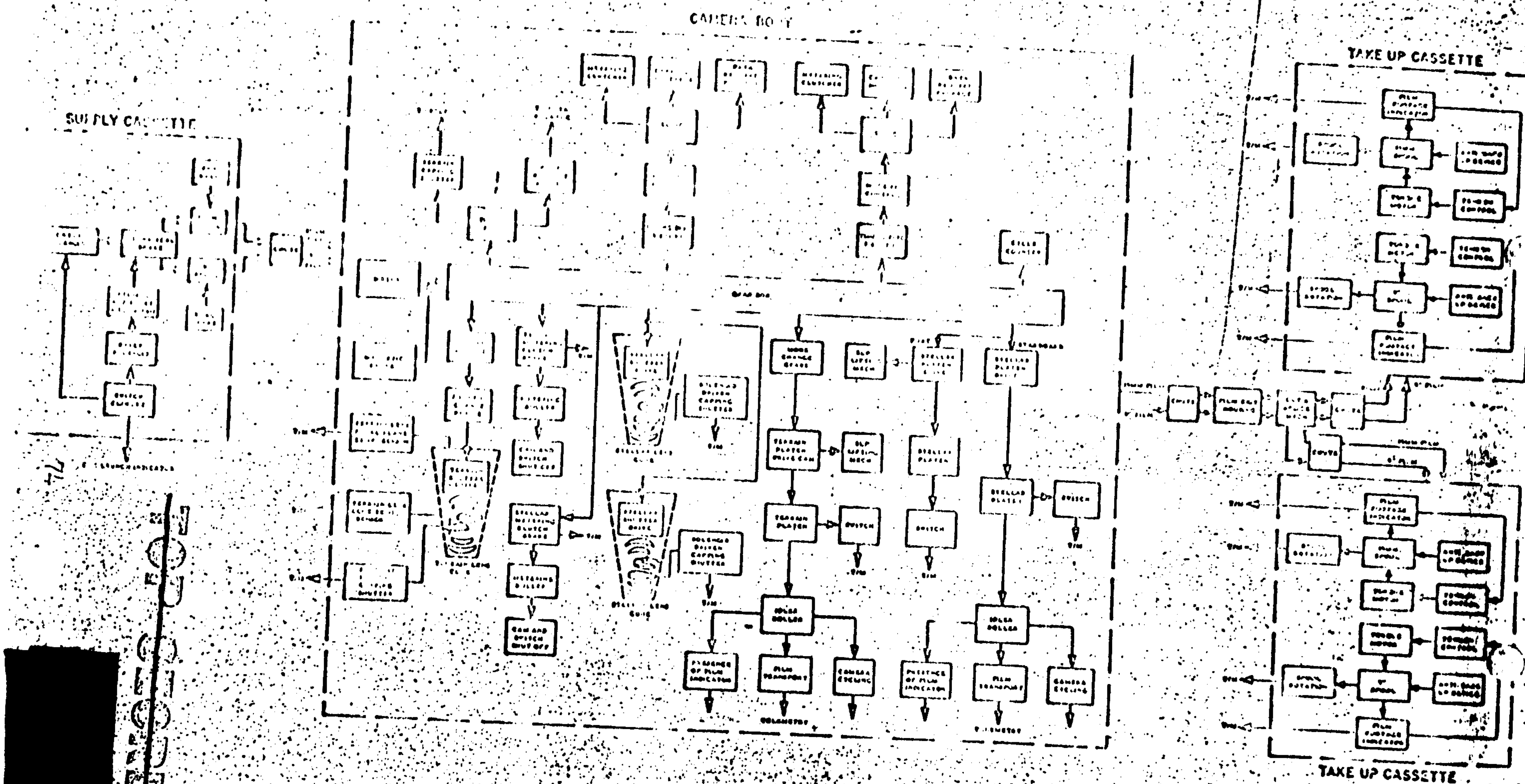


FIGURE 17

DISIC INSTALLATION-END VIEW

[redacted]

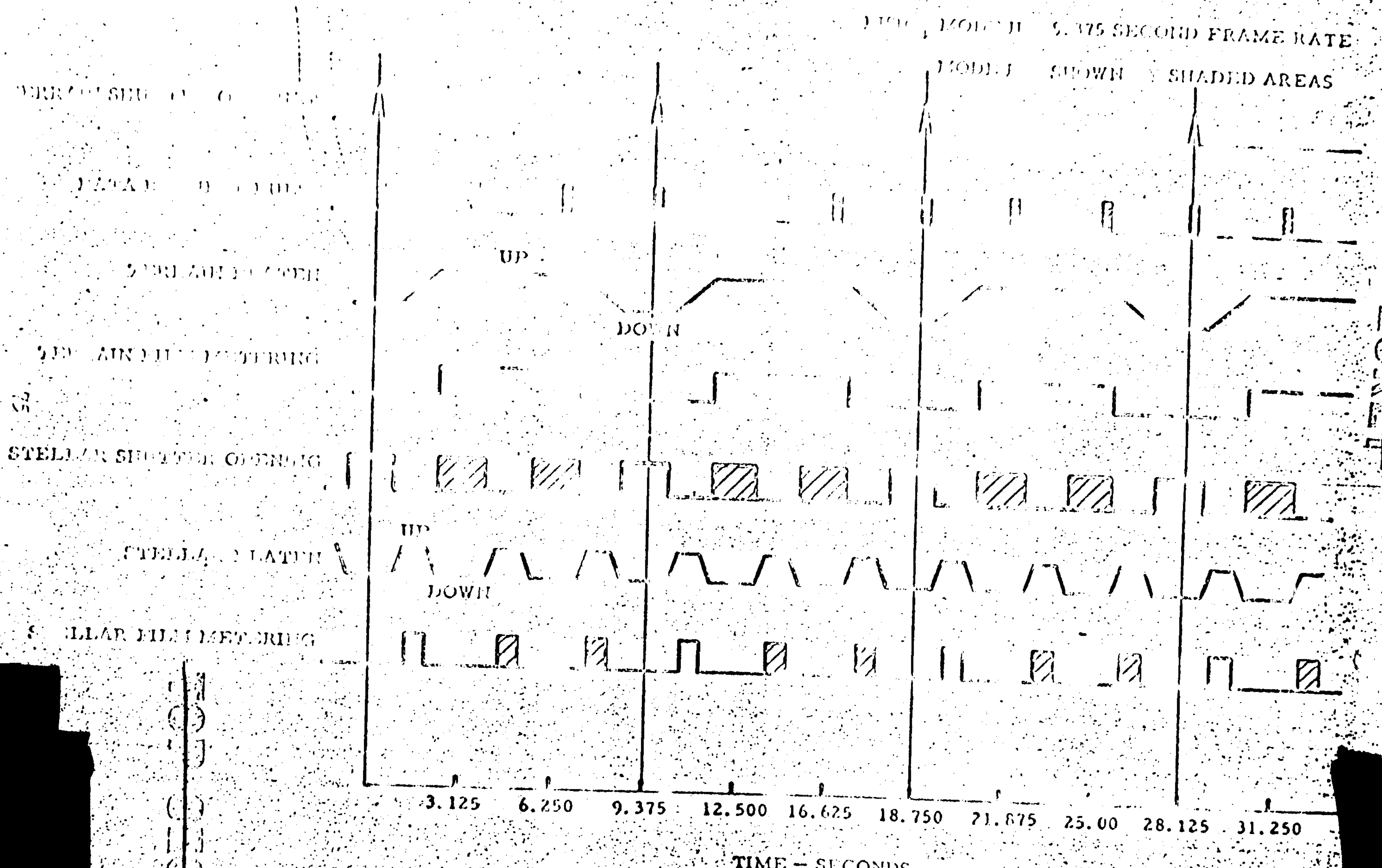


[redacted]

MECHANICAL BLOCK DIAGRAM

FIGURE 18

~~TOP SECRET~~



[redacted]

FIGURE 19

DISIC TIMING DIAGRAM

[redacted]

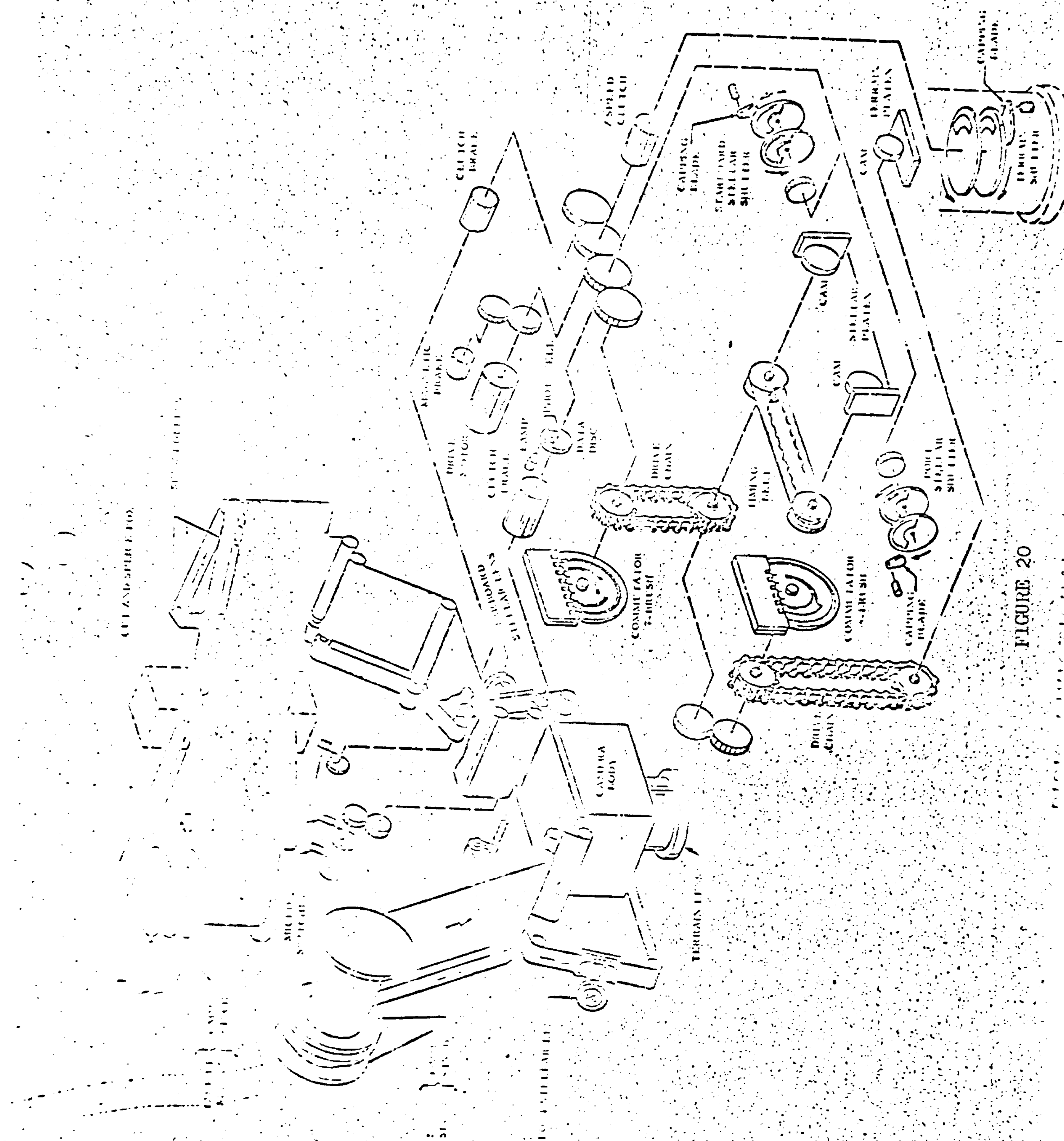


FIGURE 20

~~TOP SECRET~~

[redacted]

improve the photography. The capping shutter in either of the stellar cameras can be left in a capped position to avoid direct sunlight entering the camera. The exposure time of the terrain camera can be changed from 1/500 second for normal solar elevations to 1/250 second for low solar elevations.

The DISIC contains two film paths: a 35 mm. stellar film and a 5-inch terrain film. The 35 mm. film passes through both stellar cameras and is metered such that the imagery from each camera is interlaced as shown in Figure 21. No alteration is made to the film metering when one of the capping shutters is closed.

Camera Formats

A sketch of the stellar camera format is shown in Figure 22. The lens serial number is exposed by each stellar camera while the time word is exposed only by the port stellar camera. The port stellar image is further identified by a "P" after the lens serial number. The detail of the time word display and start-of-pass mark are shown in Figure 23. The format sketch of the terrain camera is shown in Figure 24, and the detail of the time word in Figure 25.

Camera Calibration

The DISIC reseau intersections are calibrated prior to camera assembly. DISIC subsystem calibration data are obtained through stellar photographic (static) tests and Army Map Service (AMS) performs the reduction of these data. The calibration accuracies are listed in Table VI.

DISIC CALIBRATION

PARAMETER	TERRAIN	STELLAR
Distortion:		
Radial	3 Microns	3 Microns
Tangential	3 Microns	3 Microns
Reseau Intersections	1 Micron	1 Micron
Principle Axis to Reseau Intersection	3 Microns	3 Microns
Three Angles:		
Tilt	5 Arc Seconds	5 Arc Seconds
Swing	20 Arc Seconds	20 Arc Seconds
Azimuth	30 Arc Seconds	30 Arc Seconds
Equivalent Focal Length	10 Microns	10 Microns
Calibrated Focal Length Computed to	1 Micron	1 Micron

(All values above are ± and are 1 sigma.)

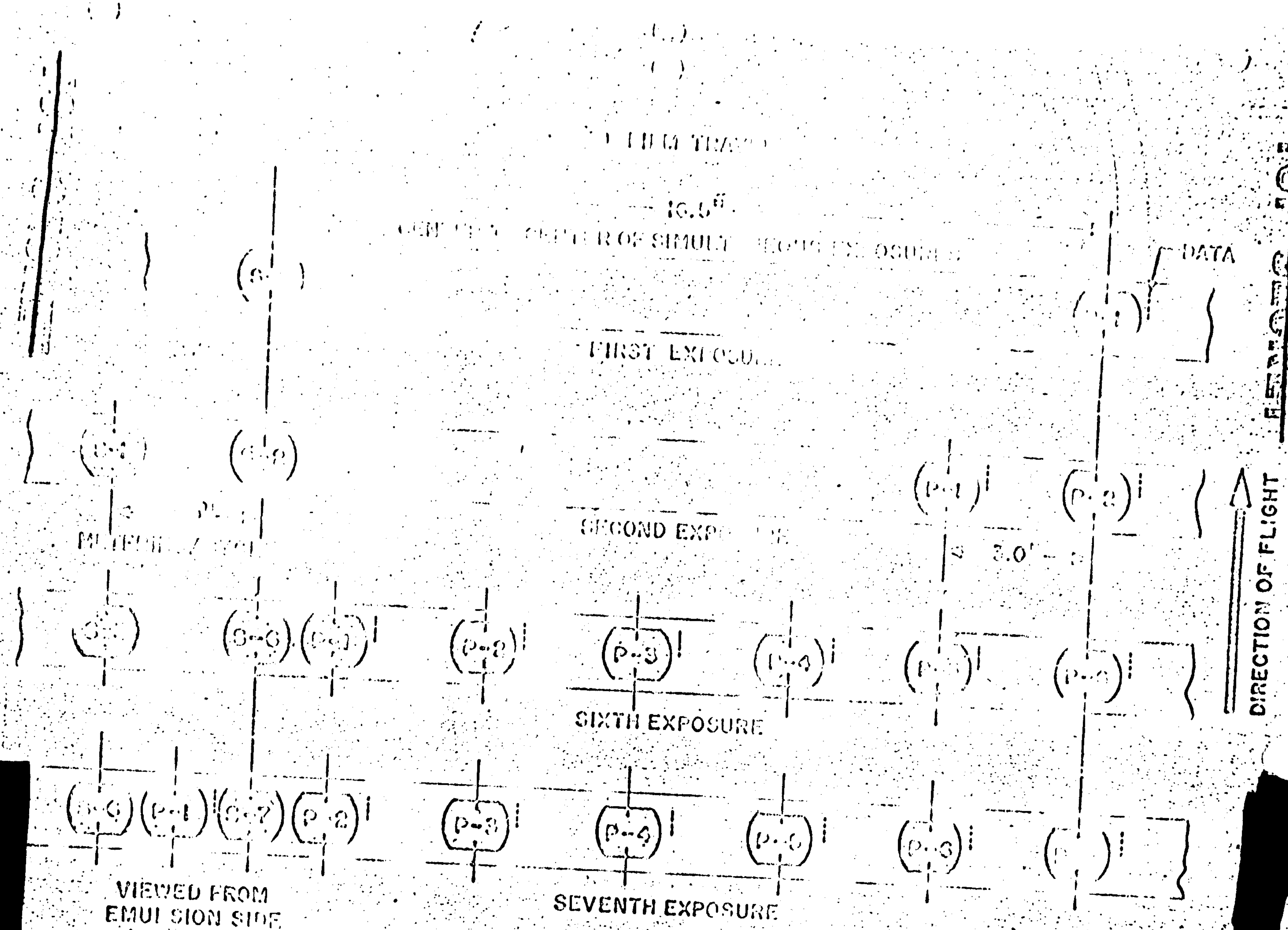
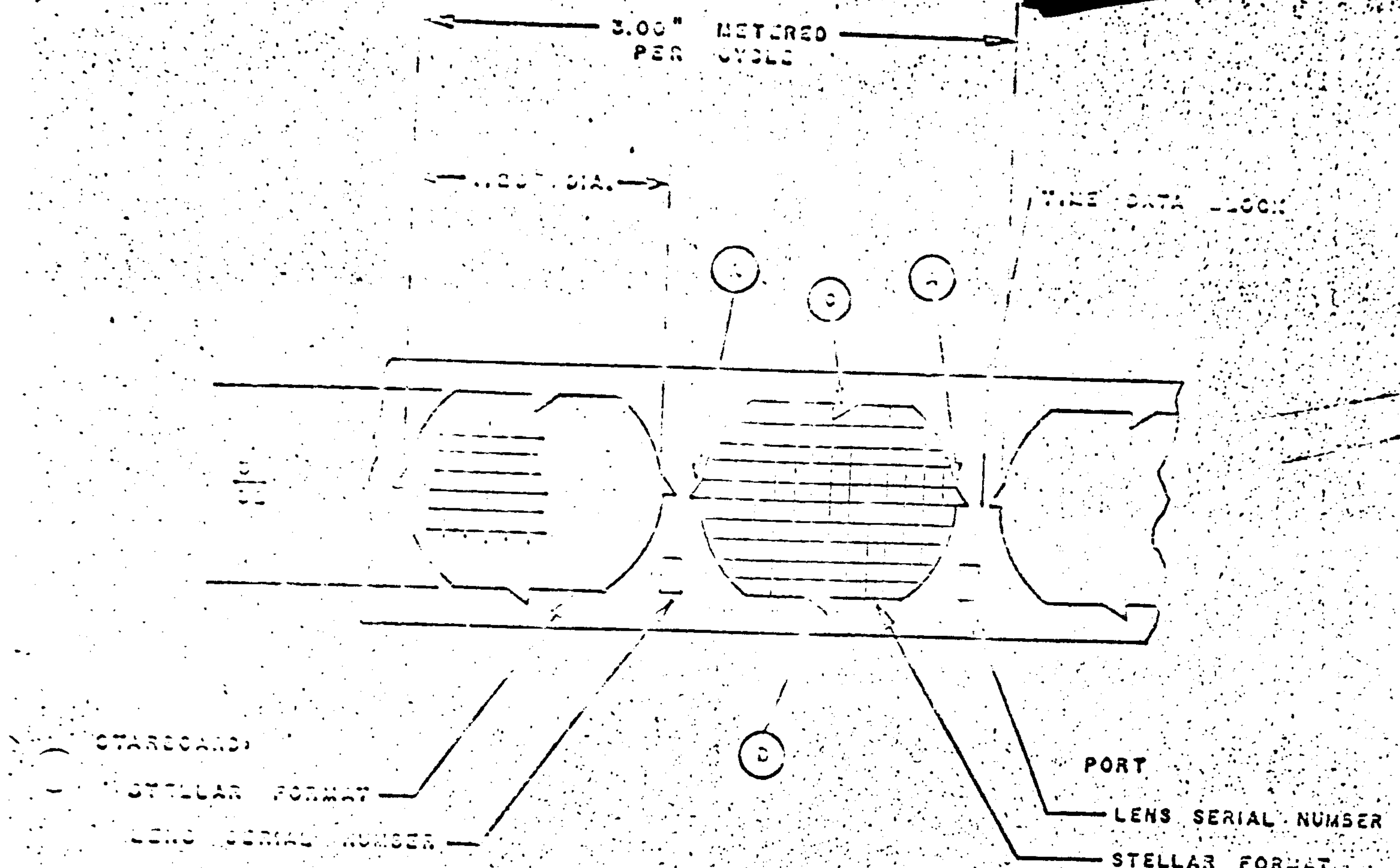


FIGURE 21
STELLAR CAMERA EXPOSURE SEQUENCE

[redacted]

[redacted]



ORIGINAL NEGATIVE
VIEWED EMULSION UP.

RESEAU GRID SPACING 2.5 MM

What does this mean?
I believe it should say that
it points to time word identified
with this frame

FIGURE 22
STELLAR CAMERA FORMAT

[redacted]

[redacted]

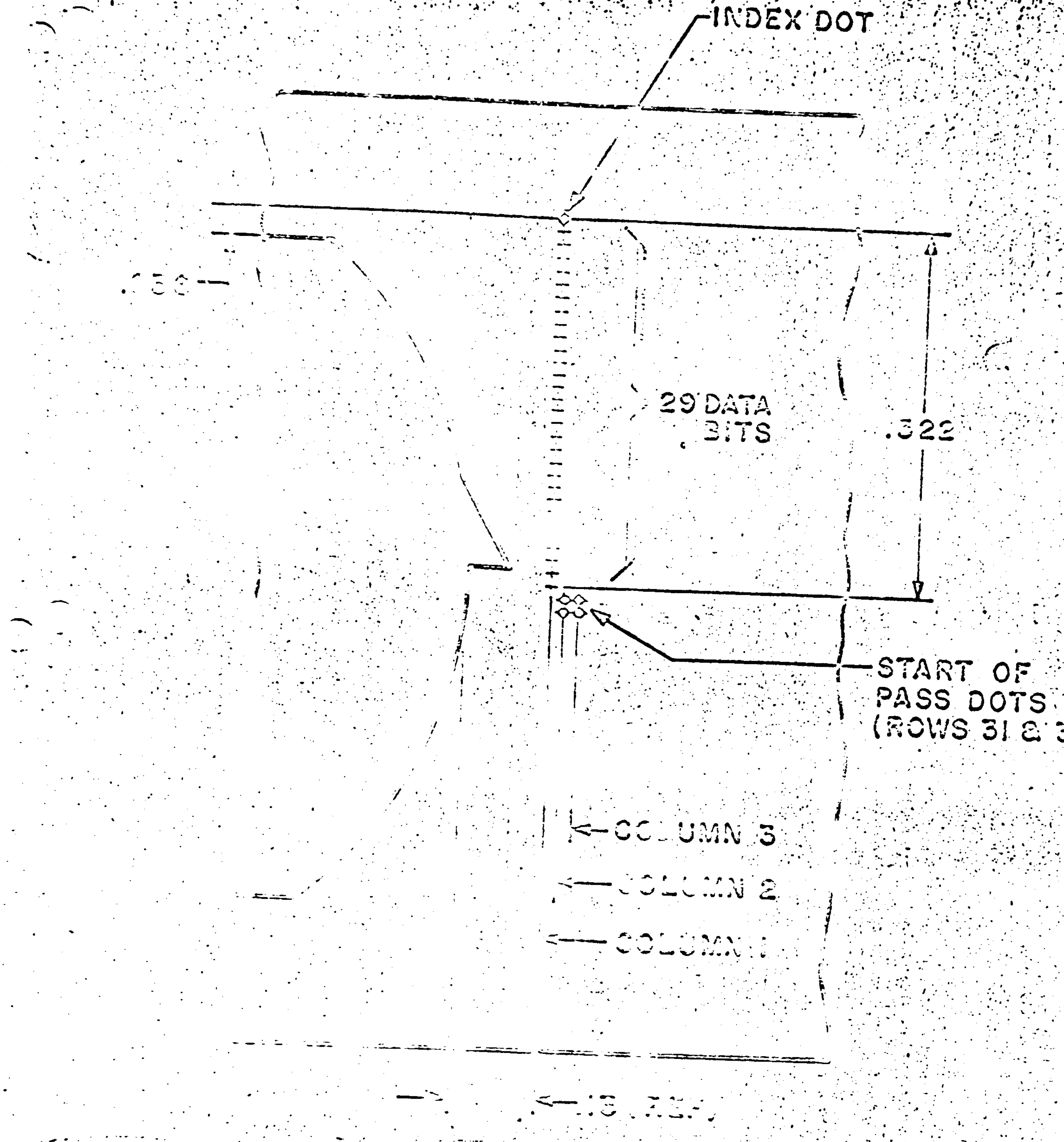


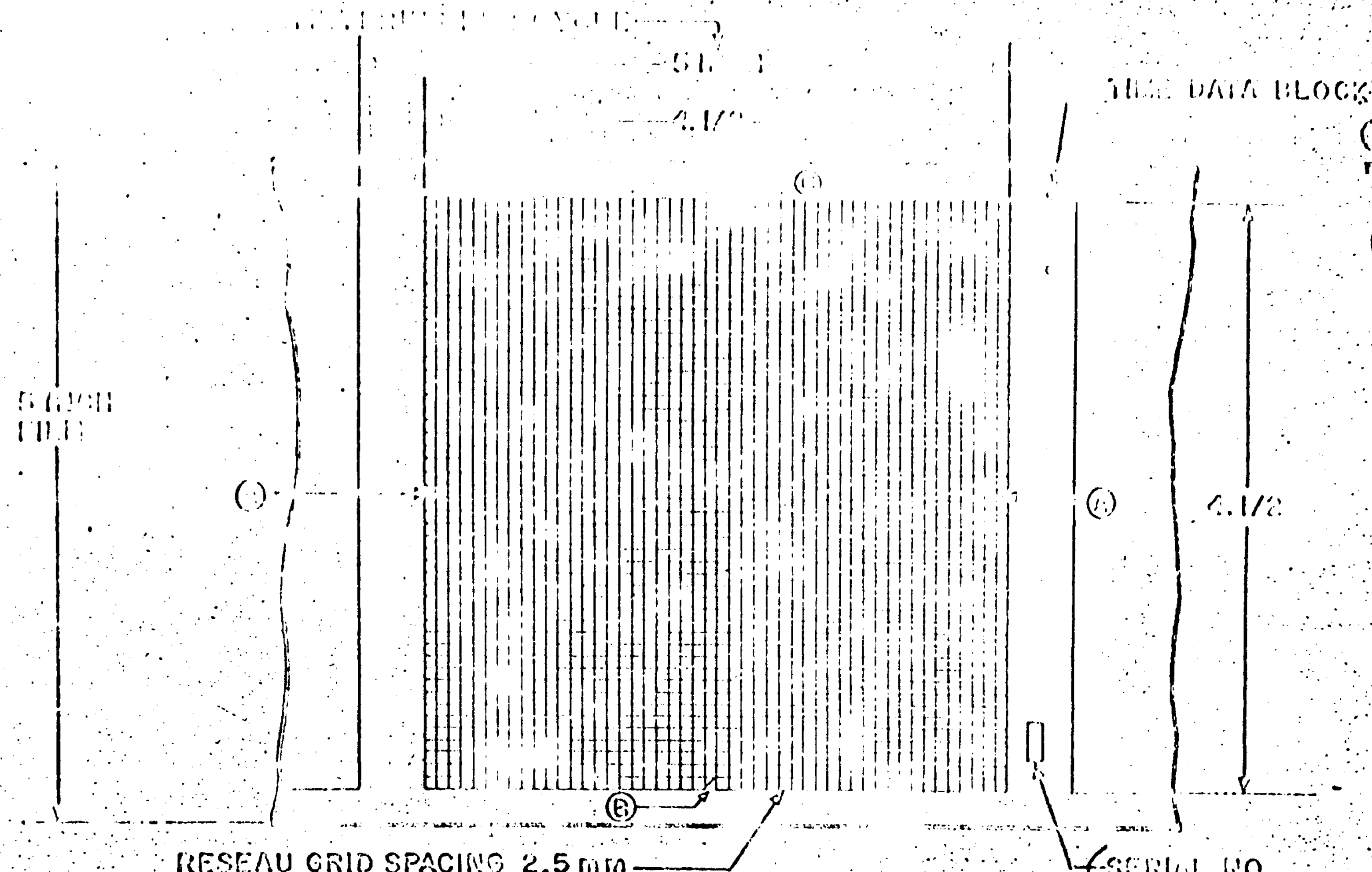
FIGURE 23

STELLAR CAMERA SL FORMAT

[redacted]

[redacted]

~~TOP SECRET~~



VIEW FROM EMULSION SIDE

[redacted]

FIDUCIAL INDICATIONS

(A) FLIGHT DIRECTION
(B) FILM METERING
(C) FRAME TIME WORD ?

FIGURE 24

[redacted]

~~TOP SECRET~~

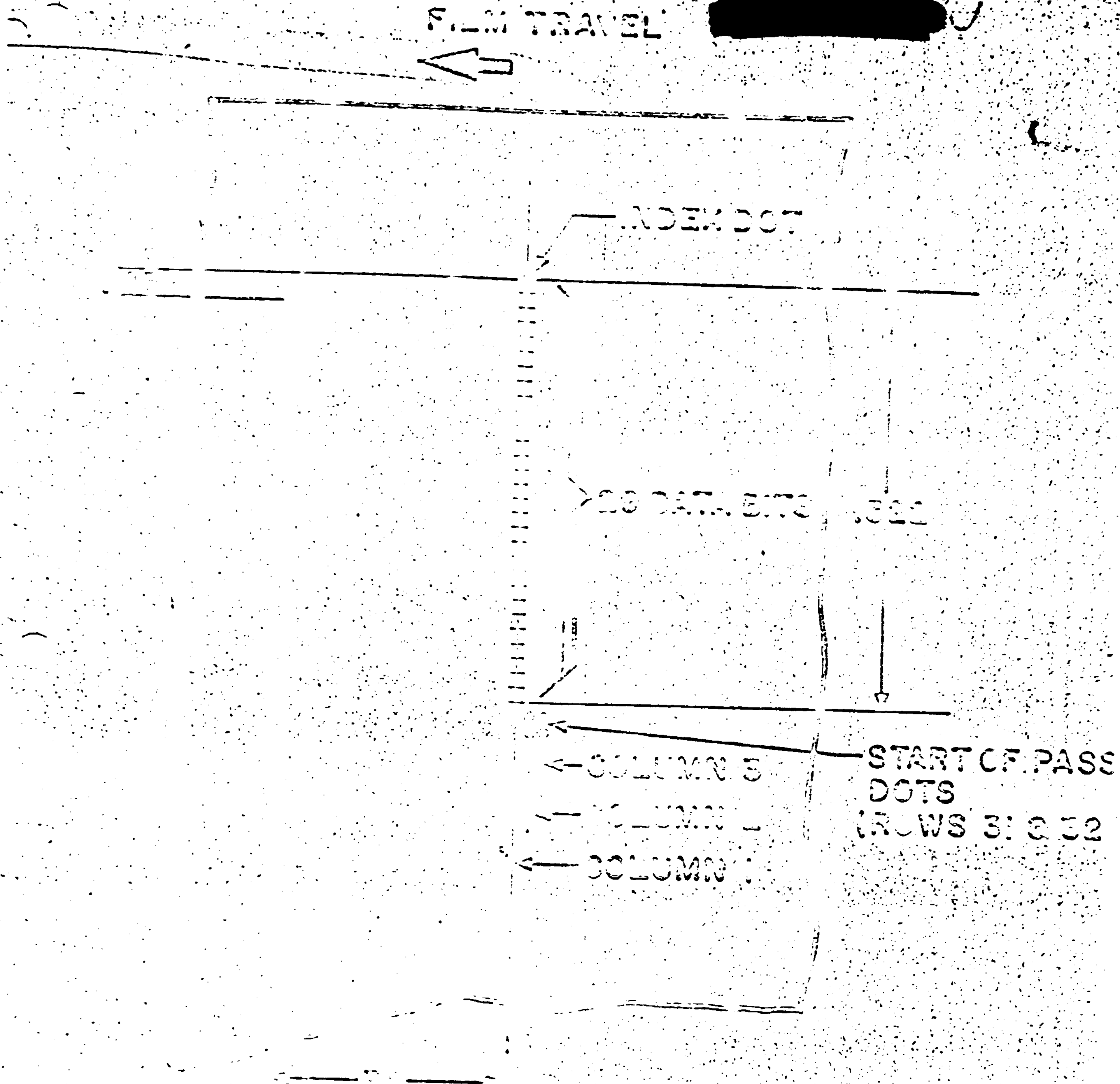


FIGURE 25
TERRAIN CAMERA CLIP FORMAT

[redacted]

Terrain Camera Coverage

[redacted]

A listing of the coverage and overlap of the terrain camera is shown below for selected altitudes between 80 and 120 N.M.

Altitude - N.M.	80	90	100	110	120
Side Dimension of Ground Pattern - N.M.	120.6	135.6	150.7	165.8	180.8
Area Coverage per Frame - Sq. N.M. x 10^4	1.45	1.84	2.27	2.75	3.27
Overlap - % 9.375 Sec/Cycle	68.0	71.8	74.4	76.7	78.7
12.50 Sec/Cycle	57.4	62.1	65.9	69.0	71.6

Telemetry

The sequence of camera functions that are instrumented for telemetry are illustrated in Figure [illegible].

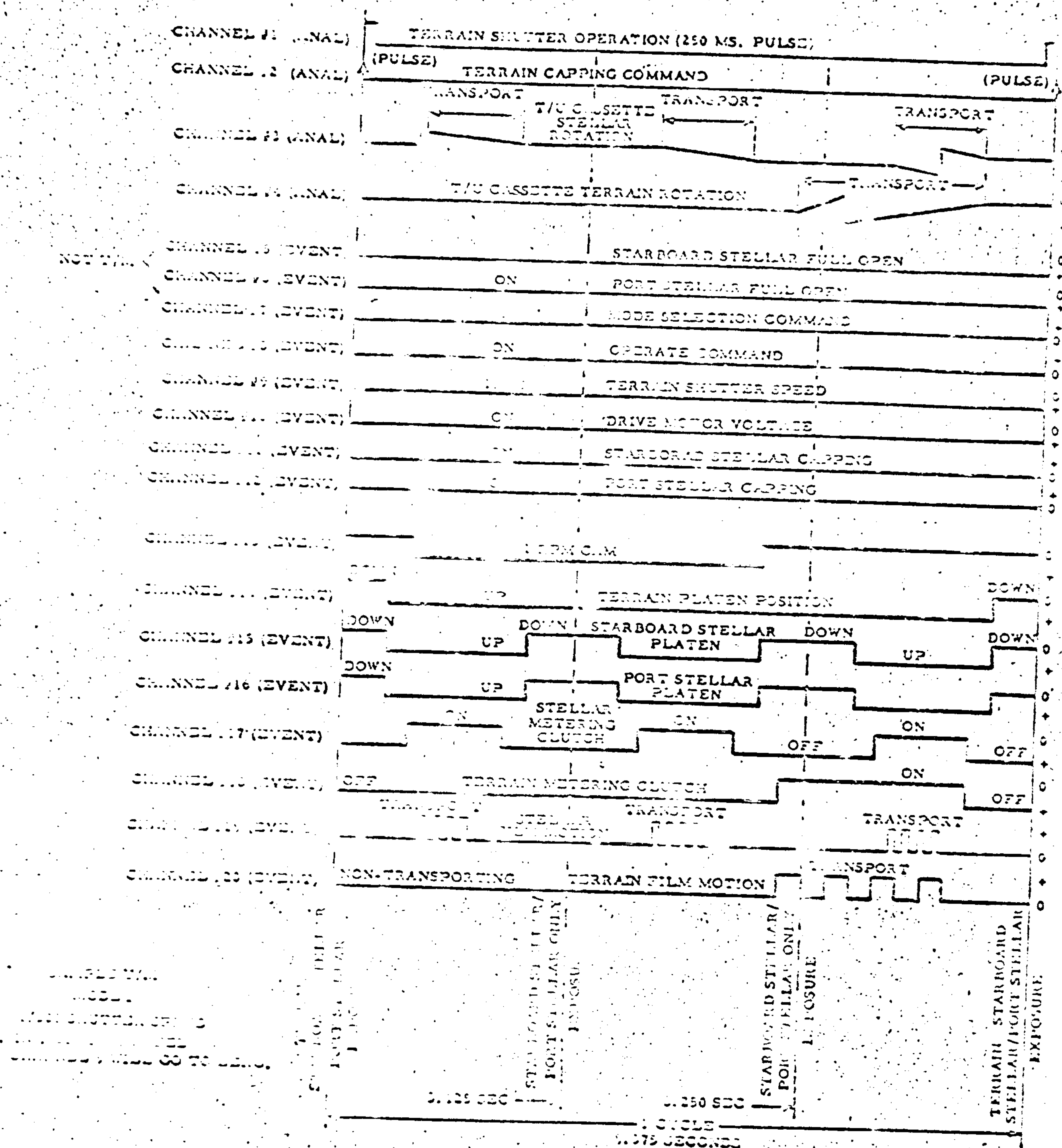
Satellite Recovery Vehicle

The basic recovery subsystem consists of two Mark 5A Satellite Recovery Vehicles mounted in tandem on the payload structure as shown on Figure 5. The primary function of the SRV is the return of payload material from orbit. This is accomplished by separation of the SRV from the satellite vehicle, de-boost from orbit, reentry and subsequent parachute deployment and ablative heat shield separation. Recovery is 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 is designed to float and to be acquired by a surface force.

Figure [illegible] illustrates the general arrangement and primary components of the basic SRV.

[redacted]

 [redacted]



TELEMETRY TIMING DIAGRAM

FIGURE 25

[redacted]

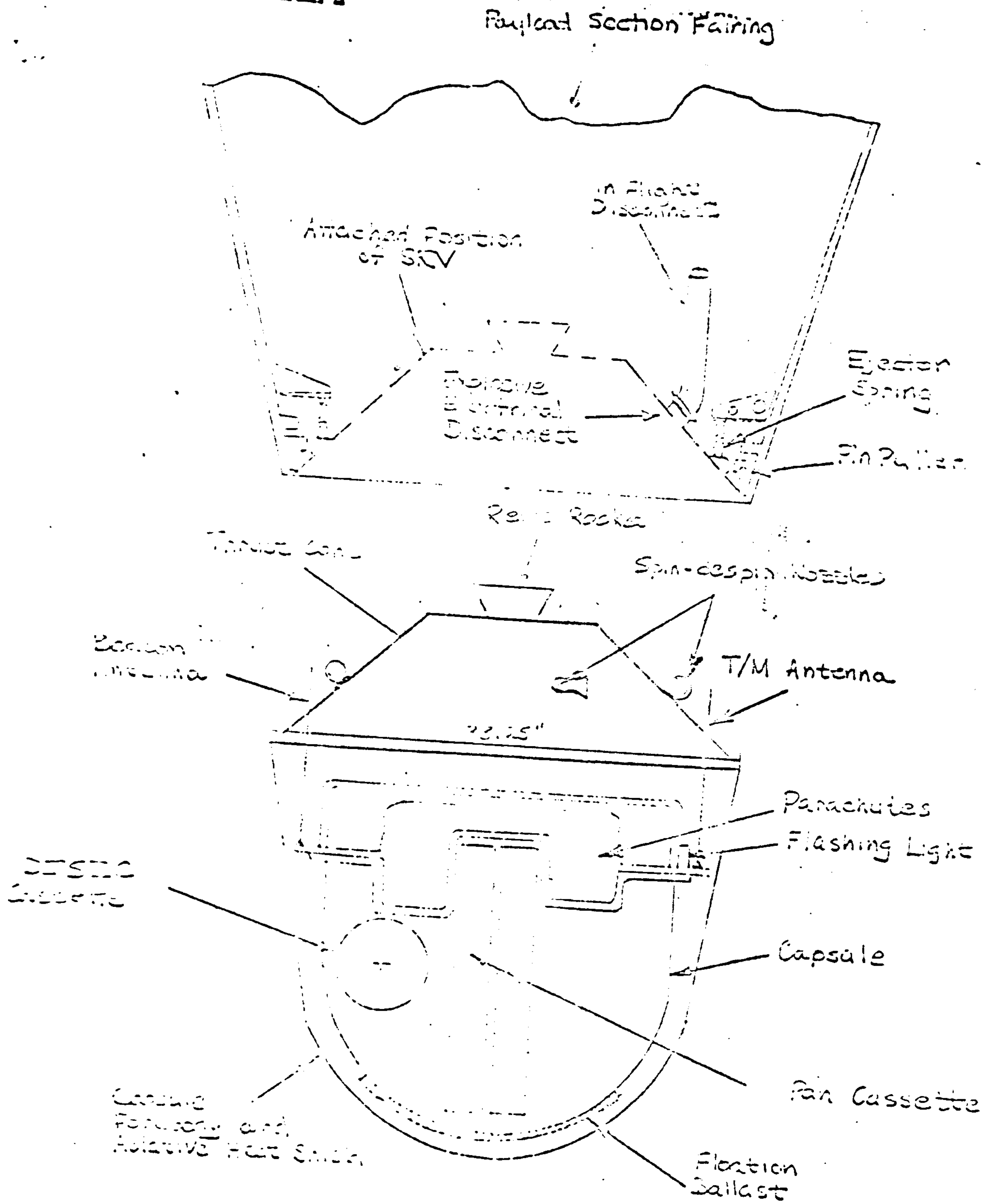


FIGURE 17 Satellite Recovery V[redacted] Profile

Structural Design

The maximum weight of the SRV is 417 pounds. Components in the capsule and on the thrust cone are packaged to provide space and access for payload components. The relative location and orientation of the pan camera and DISIC take-up cassettes are shown in Figure 21. Sealing provisions are provided for the capsule and other devices as necessary to protect the exposed film during the reentry and retrieval operations.

The capsule modifications are designed to float for at least 55 hours in the event that air retrieval is not accomplished. The sinkport supplied with the capsule scuttles the capsule after 55 hours. Flotation is designed not to exceed 85 hours maximum. The capsule is 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 floats and will not capsize in sea states of 3 or less, as defined by the U.S. Navy Hydrographic Office.

The parachute recovery subsystem effects the necessary deceleration and stabilization of the recovery capsule during descent through the atmosphere. The maximum suspended weight, excluding the parachutes, is 230 pounds. The desired rate of descent at 10,000 ft. above mean sea level is less than 30 feet per second under standard atmospheric conditions. The main parachute canopy is designed for aerial recovery, with 90 to 230 pounds suspended weight. Maximum aerial recovery altitude is 15,000 ft., and maximum aircraft speed is 135 knots indicated air speed.

Command, Control and Telemetry

All commands to initiate SRV operation are provided from the satellite vehicle. After separation from the payload, the sequence of spin, firing the de-boost rocket, despin, and ejection of the thrust cone, are controlled by a timer located on the thrust cone. After the reentry phase, deployment of the parachutes is initiated by circuitry contained within the SRV. The SRV provides a 3 channel FM/FM telemetry system for supplying key event and environmental data. A continuous wave transmitter is provided as a beacon to assist in acquiring the capsule during recovery.

Attitude Control and Propulsion

During the separation sequence, the satellite vehicle provides the initial orientation of 120 degrees from the local horizontal. Subsequently, the SRV attitude is maintained during the retro-rocket firing by spin stabilization. A spin rate is 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 is despun. During the balance of the reentry phase, SRV attitude is maintained within acceptable structural limits by the aerodynamic damping characteristics inherent in the SRV design.

The retro-rocket provides total impulse of 10,500 lb. seconds ± 3%.

Instrumentation

The SRV and payload components installed in the SRV incorporate adequate instrumentation to provide status and diagnostic data during orbital, separation and reentry phases of flight. During the orbital and separation phases, SRV-mounted payload components are monitored by means of the satellite vehicle telemetry link. During the de-boost, reentry and recovery phases, status data and key events are transmitted via SRV telemetry.

Electric Power

During on-orbit operation, power to operate the environmental heater in the "A" SRV and in both SRV's is provided by the satellite vehicle. The satellite vehicle provides power to accomplish activation of the SRV and separation from its payload mounting structure during the de-boost sequence. Subsequent operation of SRV timers, telemetry, beacon, and pyrotechnic devices are powered by batteries contained in the SRV.

Retrieval Aids

A flashing light, the VHF Beacon and the telemetry transmitter on the SRV are used for tracking during the recovery phase. The flashing light has an output of 10 lumen seconds per flash with a minimum flash rate of 60 per minute. Minimum operating time after water impact is 10 hours. The minimum life of the beacon and batteries are also 10 hours after water impact. In the event of failure of one (1) recovery battery, the remaining battery provides operation for a minimum of 5 hours. Minimum operating life of the SRV telemetry subsystem and batteries is 20 minutes after separation of the SRV from the payload, but the design goal is 40 minutes.

SECTION V

OPERATIONS

The Corona/J-3 System encompasses the total capability necessary to achieve search-surveillance photography by orbiting satellite, and includes all functional flight and ground based systems with support personnel necessary to attain this objective.

A summary of the J-3 operational data is as follows:

Active lifetime	14 days
Camera operating altitude	80-200 N.M.
Period	88-91.5 Min.
Perigee altitude	80-110 N.M.
Inclination	60°-110°
Location of perigee	20°-60° No. Lat. Descending
Beta angles	+65° to -65°
Reentry first SRV	1 to 10 days
Reentry second SRV	2 to 14 days
Panoramic Camera at 90 N.M. photographic altitude:	
Stereo angle	30°
Scan angle	70°
Field of view	5.12°
Cycle rate	1.5-4.5 Sec/Cycle
Maximum duty cycle	20 Min./Orbit

Film load per camera	15,750 Ft. / 6,000 Frames
Forward overlap	7.6%
Swath width	130.5 N.M.
Forward coverage	8.68 N.M.
Total stereo coverage	6.3×10^6 Sq. N.M.

DISIC at 90 N.M. photographic altitude:

	Terrain	Stellar
Field of view	74°	23.5°
Cycle rate		
Slave mode	9.375 Sec/Cycle	3.125 Sec/Cycle
Alt	12.5 Sec/Cycle	3.125 Sec/Cycle
Indep. mode	9.375 Sec/Cycle	9.375 Sec/Cycle
Alt	12.5 Sec/Cycle	12.5 Sec/Cycle
Maximum duty cycle	45 Min./Orbit	45 Min./Orbit
Film loading	2,000 Ft.	2,000 Ft.
	4,800 Frames	16,000 Frames
Coverage	26.3×10^6 Sq. N.M.	
Overlap	71%	

COMMUNICATIONS AND CONTROL

Communications and control are obtained through the use of the USAF Satellite Control Facility (SCF) tracking, telemetry and command net operating under the control of a centralized mission control center, the Satellite Test Center (STC), located at Sunnyvale, California. Tracking stations of the SCF that are utilized to perform this function are the [redacted] Tracking Station [redacted] the [redacted] Tracking Station [redacted], the [redacted] Tracking Station [redacted] the [redacted] Auxiliary Station [redacted] the [redacted] Tracking Station [redacted] the [redacted] Tracking Station [redacted] and the [redacted] Tracking Station. Other stations which may or may not be a part of the SCF may support operations as necessary on an individual flight or flight series basis if required. The maximum number of consecutive orbits between station contacts is four.

The SCF is 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 are also obtained by SCF stations, and are made available for reduction, analysis and display at the STC.

Payload and satellite vehicle on-orbit functions are pre-programmed on an orbital programmer tape for the desired nominal mission. Flight alternate stored programs for photographic operations are also provided for selection under non-nominal orbit conditions, predicted weather conditions, and variations in mission priorities. Adjustment of the orbital programmer, the selection of stored programs and camera operating functions are provided through real-time commands based on actual ephemeris conditions as determined from tracking data. Tracking and commanding capabilities are adequate to compensate for the following effects:

A. Accuracy tolerances on the orbital injection vector affecting orbital period, eccentricity, and location of perigee.

B. Orbit decay with time due to atmospheric drag effects on the satellite vehicle.

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

To synchronize the stored commands with vehicle position, the orbital programmer is adjusted by real-time command so that the in-track error does not exceed 3.5 seconds of time. Accurate ephemeris data is also provided for real-time selection of image motion compensation and camera cycle rate, camera operating program and camera operating mode. Primary data for ephemeris prediction is obtained by the use of a PRELORT radar and a satellite-borne transponder.

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

The primary command system for the J-3 System is the S-Band radar analog command mode, which comprises 6 tones and 15 commands. Two additional command systems are used to augment the primary command capability. These are the ZEKE (150 mc VHF) and ZORRO (S-Band digital). Depending upon the command, ZEKE commands require transmission in a secure (encoded) or a clear mode. ZORRO commands are transmitted in a secure mode. Antennas used by SCF ground stations to transmit commands to the satellite vehicle are as follows:

Station	Command System		
	ANALOG	ZORRO	ZEKE
[redacted]	Prelort	Prelort	Helix
[redacted]	Prelort	Prelort	Helix

[redacted]	Prelort	Prelort	Crossed-Yagi
[redacted]	Prelort	Prelort	Helix
[redacted]	Prelort	Prelort	Crossed-Yagi
[redacted]	Prelort	Prelort	Crossed-Yagi

At appropriate times following completion of the orbital missions, the reentry vehicles are separated and ejected from orbit using their own de-boost propulsion capability to impact in the selected retrieval area. The primary impact area is a broad ocean area within the WTR, located between 10° and 26° N. Latitude, and 145° to 172° W. Longitude. The nominal impact latitude is 24° N. Retrieval is accomplished by air recovery as the primary mode, or by water recovery in the event that air retrieval is not accomplished. The STC computes impact predictions for use in commanding de-boost of the SRV's, and for deployment planning by the recovery forces.

Under normal operating conditions, vehicle and payload commands specified for the flight are 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 Payload Sub-Assembly Project Office (PSAPO) for the payload. In the case of abnormal flight conditions or anomalies of the vehicle and/or payload, commands are subject to review, approval and control of the [redacted]gram Directorate.

Vehicle commands transmitted by the SCF are based upon ephemeris data obtained and reduced by equipment presently in use in the SCF. Payload command is based on the payload data available generated by the PSAPO.

~~TOP SECRET~~

[redacted]

PRE-MISSION OPERATIONS

A mission is scheduled by the [redacted] approximately 30 days prior to launch. The [redacted] provides a list of inclination angles, that can be flown, to the [redacted] usually four or five in number. Along with each case is an indication of the weight limitations and the possible days on orbit for each. [redacted] examines each of the cases and evaluates each against the Intelligence Requirements, the forecast weather conditions, the area coverage capability, and the sun angle characteristics of each. The inclination angle selected is then sent to [redacted]

In addition to the inclination angle selection, the launch window must be selected. To do this, [redacted] ceives the launch window limits for the vehicle from [redacted] nd the launch window limitations on the payload (pan camera and stellar index camera) from APF (Advanced Projects Facilities, Palo Alto). [redacted] then reviews these and picks the optimum (best lighting condition) and specifies the launch window and the recommended launch time.

After the orbit and launch time have been established, [redacted] is responsible for generating the camera program options to be used for the flight. The CORONA system requires all the possible payload command options be loaded into the vehicle prior to launch.

The CORONA targeting requirements are specified to [redacted] by the Photo Working Group Committee of COMIREX. It includes the priorities for the area search requirements and priorities for mapping/charting holiday areas as provided by DIA through the Army Map Service. The target deck includes the high priority search complexes (usually 15-50 in number). These are designated as priority 1's and the balance of the target deck is made a lower priority.

There are nine different camera program options that can be specified. The basic objective used in selecting the program options is to plan the camera on-off sequencing in the different programs to provide optimum flexibility on each orbit. In order to support the [redacted] in defining the camera program options, the Central Intelligence Agency (CIA) provides computer support to the [redacted] for this initial programming. They provide to [redacted] and APF a mission profile entitled GENIE, a program listing (camera off and on) and a target acquisition list. This information is transmitted to the [redacted] at launch minus 9 days, at which time [redacted] edits the camera program (as generated by the CIA computer) for intelligence coverage, and transmits the program changes to APF. APF checks the original program and the added changes to insure compatibility with vehicle capability. If any changes are required, APF will transmit these to the [redacted] for their review. At launch minus 3 days, APF cuts the camera program tape and transmits a mission program consumption listing for each possible camera operation to [redacted] Weather Central, and the CIA.

ON ORBIT OPERATIONS

The [redacted] is responsible for selecting the CORONA camera program option for each rev. The object in this selection is to effectively manage the mission film usage in order to optimize intelligence collection over the anticipated days on orbit. Prior to each vehicle load, the [redacted] receives a weather forecast for the revs under consideration. This information, along with current mission coverage, is used to assist in selecting the program option that will allow the camera to operate only over those areas for which photographs are desired. Once the program option has been selected, the [redacted] sends the message to APF and the STC. The APF translates the message into vehicle language, and the STC has the primary responsibility for transmitting the commands to the vehicle.

LAUNCH OPERATIONS

Program [redacted] vehicles are launched from Vandenberg AFB. Launch operations are under the cognizance of the 6595th Aerospace Test Wing. The following launch facilities are used to assemble, check out, and launch the SLV-2G/SS-01B boost vehicle:

Space Launch Complex (SLC)-1	East Pad
Space Launch Complex (SLC)-3	West Pad

Program [redacted] operations are supported by the Western Test Range (WTR) 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 is made available for collection of down-range telemetry data which is not within reception range of a land-based station. The maximum launch rate is two per calendar month, with an expected ~~normal~~ launch rate of SEVEN TO ten per year. A turn-around time of 14 days from launch to launch for any given launch pad is desired. Additionally, a provisional capability exists to hold a Program [redacted] launch vehicle in a state of readiness for extended periods of time, up to 20 days, with accepted degradation in performance and reliability so that launch can be accomplished within 24 hours when so directed. The system goal for successfully accomplishing the initial countdown and launch within the window is a 70% probability of success. The mission to be flown by this standby vehicle remains fixed throughout the standby period. When this requirement is in effect, maintenance for this particular vehicle and ground equipment is scheduled and performed in such a manner as to not invalidate the ability to launch within 24 hours. If the situation arises that this vehicle must be demated, an alternate vehicle and launch pad is phased into the standby status, provided that the particular requirement is still in effect. Although the standby vehicle is given highest priority at the time its launch is directed, the capability to check out and launch Program [redacted] vehicles from the remaining assigned pad is not impaired during the standby period.

Final checkout, loading, and mating of the payload equipment to the satellite vehicle is performed under conditions of the strictest security. Appropriate facilities and personnel are 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.

ON ORBIT OPERATIONS

The [redacted] is responsible for selecting the CORONA camera program option for each rev. The object in this selection is to effectively manage the film usage. Prior to each vehicle load, the [redacted] receives a weather forecast for the revs under consideration. This information, along with current mission coverage, is used to assist in selecting the program option that will allow the camera, as much as possible, to operate only over those areas for which photographs are desired. Once the program option has been selected, the [redacted] sends the message to APF and the STC. The APF has the primary responsibility for translating the message into vehicle language, and the STC has the primary responsibility for transmitting the commands to the vehicle although as a safety precaution they check each other.

RECOVERY OPERATIONS

The satellite vehicle recovery subsystem provides a capability for recovery on any day following liftoff. The command to initiate recovery is given from stations of the SCF.

The recovery sequence is divided into two phases: reentry and recovery. The reentry phase starts with the initiation of a programmed command. Following this command, the SRV beacon and telemetry is turned on to permit detection, tracking and data recording of the reentry sequence.

The satellite vehicle is pitched over a nominal 120 degrees from the local horizontal, the film cut, the SRV sealed, and the SRV separated from the satellite vehicle. De-boost is achieved by means of spin jets, a retro-rocket, and de-spin jets. Immediately after de-spin, the thrust cone (mounting platform for the rocket and spin system), is separated from the reentry vehicle. The recovery phase consists of the deployment of the parachute system, ejection of the ablative shield, and activation of a flashing light at approximately 50,000 to 60,000 feet.

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

Two SRV's are carried by each satellite vehicle. Recovery of the first SRV is accomplished in from one to ten (10) days after launch and the second SRV is recovered in from two to fourteen (14) 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 is normally of seven days duration, however, a capability exists to cut the film in the "A" SRV and continue the mission in the "B" mode with subsequent "A" recovery at a later time.

Communications channels exist between all air and surface units of the recovery forces. Communications between the units of the recovery forces and the STC are 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 are under the jurisdiction of the AFSCF Recovery Control Group (RCG), Honolulu, Hawaii. Logistic support is 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 is in accordance with the directives of the Special Projects Directorate (SPD). Until such time as the designated courier is able to assume physical custody, the Commander, AFSCF-RCC is responsible for the capsule's physical and security safeguarding per designation by the Commander AFSCF. Upon assuming physical custody, the courier is responsible for capsule handling and security during transportation and delivery to the designated processing center.

MISSION CHARACTERISTICS

Launch Reaction Time

Launch Reaction Time is defined as the time span necessary to complete all pre-launch preparations and accomplish the launch, starting from the time a particular mission is defined by the Special Projects Directorate. For the SLV-2G/SS-01B vehicles the following items require hardware setting or other action between R-8 and launch based on mission peculiarities and the launch pad used.

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.

Preparatory work to support readiness of the above items involves trajectory computations and data exchanges between participating contractors which normally require lead times from launch of 8 days. This assumes [redacted] as the standard aft payload configuration. Any special research payload may require additional programming preparation.

Ascent

Program [redacted]unches using the SLV-2G/SS-01B vehicles will be conducted from VAFB. WTR facilities are used for tracking, telemetry, range safety, and range frequency interference control.

The launch azimuth is compatible ...h orbit incl... ...ion req...rements and range safety restrictions. For i... ...ions bel... ...proximately 3 degrees, a yaw (dog-leg) maneuver is ...ired be... ...of range ... limitations on launch azimuth. Hence ...r traje... ...es which require orbit inclination angles of less tha... degrees ...e dog-leg maneuver is accomplished aft... ...ange ... has passed the critical range safety boundary.

The launch window is normally not less than pl... ...d minus one half hour about the optimum launch time. The optimum launch time is computed for each flight on the basis of required ground search area lighting conditions and vehicle thermal considerations. Within the above constraints, the launch time and window may be varied to obtain the best thermal environment in-orbit for payload, and satellite vehicle temperature-sensitive equipment and horizon sensor ascent look angle.

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

Event	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	
Stage II Engine Ignition	240.5	170.5
Solid Motor Impact	369.76	21.78
Booster Impact	642.8	813.85
Stage II Engine Cutoff (orbit injection)	483	869

<u>Orbital Elements</u>

For a particular flight the orbit parameters are specified on the basis of payload search area considerations and performance available from the launch vehicle subsystem. Parameters of primary importance are the orbital period, perigee altitude, location of perigee and orbital inclination usually in the aforementioned order. ~~To cause the satellite vehicle to overfly the desired ground-track with specified synchronization,~~ The appropriate orbital period is generally attained by selection of the proper orbit eccentricity.

With the SLV-2G/SS-01B Booster vehicles, the system is capable of a range of missions with orbital parameters within the following limits:

1. Range of <u>orbit inclinations</u>: 60° to 140°
   (Most probable inclinations: 65° to 110°)
2. Range of <u>perigee altitude</u>: 80 to 200 N.M.
   (Most probable perigee altitude: 80 to 110 N.M.)
3. Range of <u>orbital period</u>: 88 to 91.5 Min.
4. Range of <u>perigee location</u>: 90° N. to 90° S. Latitude
   (Most probable perigee location: 20° N. to 60° N. Latitude)

Vehicle structural limitations may preclude flying all possible combinations of the above parameters. Hence, at inclinations below 88 degrees, the injection altitude is the lowest value compatible with SLV-2G/SS-01B structural capabilities. Mission duration is dependent upon the orbit inclination and perigee altitude selected.

Nominal mission duration is 14 days of active operation. For any particular mission, the number of days duration is compatible with the orbit parameters required and the system orbital weight capability. Estimated mission duration capabilities as a function of orbital parameters for the J-3 System configuration are presented in Figure 28. Figure 29 is a typical preliminary performance capability chart using data obtained from Figure 28.

Drag Make-Up

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. For inclinations between 70 and 91 degrees, mission requirements can be satisfied by flying the longer period orbits of approximately 91 minutes (Westward closure) without a need for sustaining the satellite vehicle's orbital velocity. The eccentricity of these orbits tends to minimize the effects of atmospheric drag while maintaining an acceptable perigee altitude and location for payload operation. The operating regime of inclinations greater than 91 degrees with near-circular

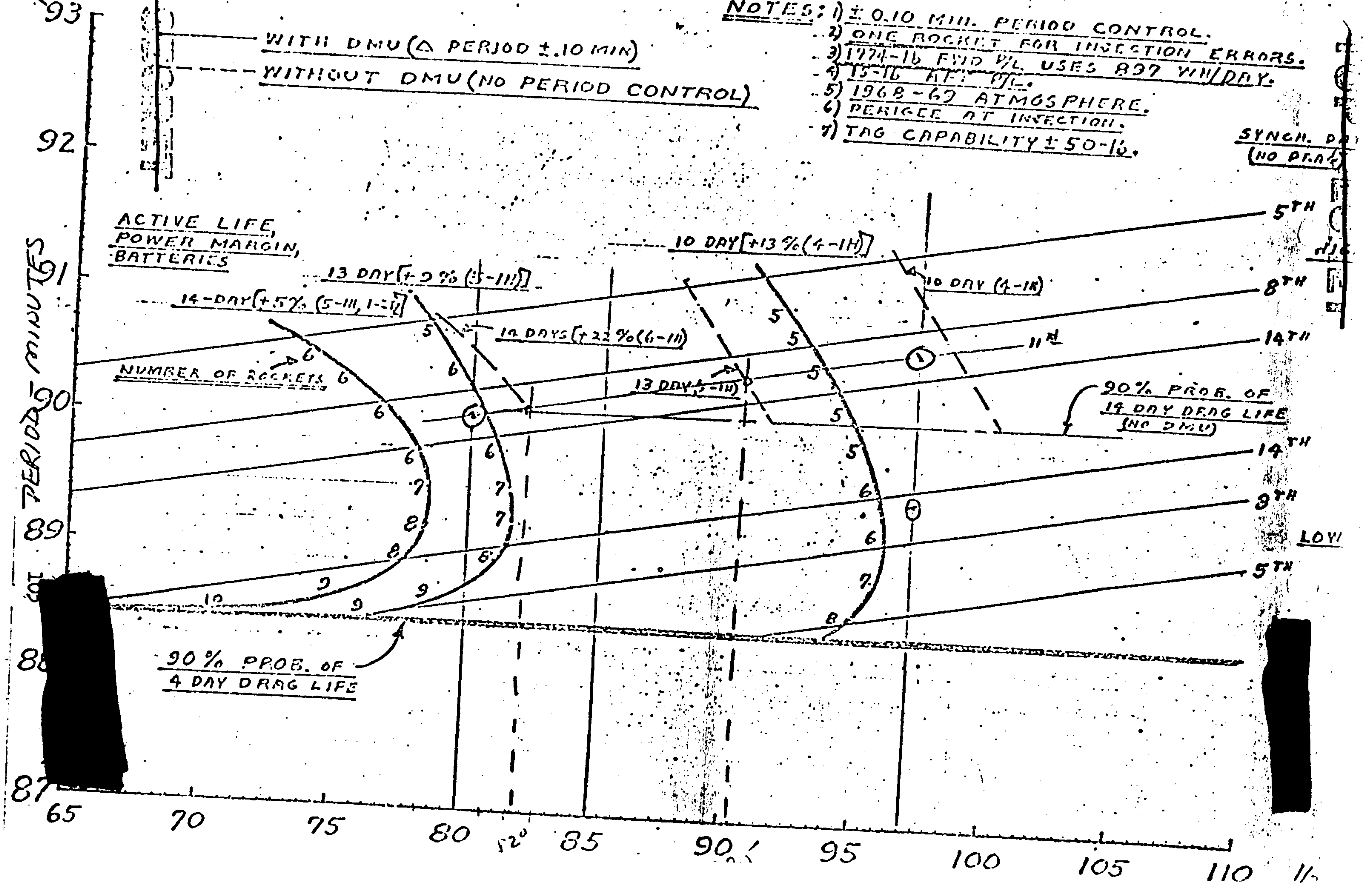
ESTIMATED MISSION CAPABILITY

THORAD/018

85 N.M. PERIGEE

1:7/67

NOTES:
1) ± 0.10 MIN. PERIOD CONTROL.
2) ONE ROCKET FOR INJECTION ERRORS.
3) 1774-lb FWD P/L USES 897 WH/DAY.
4) 15-lb AFT P/L.
5) 1968-69 ATMOSPHERE.
6) PERIGEE AT INJECTION.
7) TAG CAPABILITY ± 50-lb.



[redacted]
[redacted]

~~TOP SECRET~~ [redacted]

PRELIMINARY PERFORMANCE CAPABILITY

FTV 1641

Inclination Angle (Deg)	80		80	85	85
Launch Azimuth (Deg)	179				179
Period (Minutes)	88.82		89.95	89.01	90.13
Perigee Alt. (N.M.)	85				85
Latitude of Perigee (Deg No.)	Inject				Inject
Bottle Drop (Sec)	102				102
Synchronous Day	11th (L)		11th (H)	11th (L)	11th (H)
Aft Payload (lbs.)	15				15
DMU Rockets	8		6	7	5
Active Days	13	14	13	12	12
Performance Margin (See Note 1)(Lbs.-EWO)	-2	-26	+3	-1	+4
Power Margin (± 3%)	+9%	+6%	+11%	+10%	+6%
Batteries	4-1H 1-1C	5-1H 1-VI	5-1H	3-1H 1-1C 1-1D	4-1H 3-VI
Est. Drag Life (Stable Veh) Without DMU	6	16	16	8	18
Window-Capability With DMU	19	29	29	20	30

NOTE: 1) Performance capability based on nominal with performance margin tolerance ± 50.
2) [illegible]

FIGURE 29

~~TOP SECRET~~ [redacted]

(low eccentricity) orbits of approximately 100 nautical miles offers payload advantages of improved scale, more constant compensation of image motion, and increased opportunities for payload operation on Northbound as well as Southbound passes. The desired synchronization can be attained by flying the shorter period orbits (Eastward closure). However, a satellite vehicle orbiting at these lower altitudes is noticeably affected by the atmospheric environment and may require an orbit sustenance capability to make up the velocity decrement caused by drag. Requirements for drag make-up provisions assure a 90.0 percent probability that the satellite vehicle altitude does not decrease below 80 nautical miles during the active mission phases.

The orbit adjust uses small solid rocket motors that may be fired in the boost or de-boost direction. To achieve a de-boost capability, the vehicle is pitched to align the thrust axis in the required direction. These rocket motors have a nominal impulse of 2250 pound second per rocket.

<u>Orbit Environment</u>

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

A.	Vacuum:	1×10^{-8} 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:	35% of the solar energy (nominal)
E.	Magnetic field:	560 milliguass at the poles to 280 mg at equator for an altitude of 125 N.M.

F. Atmospheric density: The Lockheed-Jacchia or the LDENSITY atmosphere models shall be used.

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

Ephemeris

Ephemerides are 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 is used for computation of acquisition data, optimization of the mission, selection of real-time commands for payload functions, impact prediction, post-flight data correlation, flight evaluation and other such activities coincident with adequate program support. Ephemeris prediction capability accounts for such factors as geopotential harmonics through the fourth order and a seventh parameter fit for average drag determination. Satellite spatial position errors on-orbit for photographic command selection are known, after sufficient target acquisitions, to:

A. ±4.0 N.M. in-track

B. ±1.0 N.M. cross-track

C. ±0.5 N.M. in-altitude

De-boost and Reentry

De-boost and reentry is 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 is effected on North to South passes only, with South to North passes utilized for emergency. A backup attitude control capability is provided in the satellite vehicle which will allow at least one recovery on the North to South passes if the primary subsystem should fail.

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

Reentry impact dispersions are influenced by the following primary error sources:

1. Satellite vehicle attitude and attitude rates at reentry vehicle separation.

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

3. Reentry vehicle static and dynamic balance.

4. Retro-rocket impulse tolerance.

5. Uncertainty of orbit parameters at time of reentry vehicle separation.

6. Event timing errors.

7. Uncertainties in actual ballistic parameter, atmospheric density and surface winds.

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

Orbit Inclination	85°
Eccentricity	0.022
Period	91.06 Min.

Guidance & Control Subsystem	Direction	Dispersions (N.M.) Up-Range	Down-Range	Cross-Range
Primary	N to S	73	102	± 10
Primary	S to N	141	200	± 16
Back-Up	N to S	104	176	± 13

<u>Abort</u>

Abort of the launch or the orbital phases of the mission does not cause the nature of mission objectives to be revealed to unauthorized persons. In the event of an aborted launch, provisions are made to cover payload equipment under appropriate security conditions. Similarly, in the case of a catastrophic malfunction during booster ascent, a strict accounting is made of payload equipment salvage and/or disposition. In the event of an orbital phase abort, the payload is recovered, if possible. If recovery cannot be effected, a self-contained SRV timer is started with the receipt of the first recovery command (ARM). The normal reverse thrust is not performed, but recovery functions such as thrust cone eject, ablative shell off and parachute deployment are initiated at a later time than would be performed in a normal recovery. Since these events occur in-orbit, the subsequent SRV reentry due to drag ~~shall~~ will cause the SRV (with payload) to break up.

ABBREVIATIONS [redacted]

A.C.	Alternating Current
AFSCF	Air Force Satellite Control Facility
AMS	Army Map Service
ARM	First Recovery Command
Att	Attitude
Aux	Auxiliary
BTL	Bell Telephone Laboratories
cps	Cycles Per Second
D.C.	Direct Current
deg	Degree
DISIC	Dual Stellar Index Camera
DMS	Drag Make-up System
DMU	Drag Make-up
EK	Eastman Kodak
FM	Frequency Modulation
FMC	Forward Motion Compensation
ft	Feet
[redacted]	[redacted]
h.	Altitude
H.O.	Horizon Optics
[redacted]	[redacted]
IF	Intermediate Frequency
IMC	Image Motion Compensation
IRFNA	Inhibited Red Fuming Nitric Acid
IRIG	Inter Range Instrumentation Group
[redacted]	[redacted]
L/B	Lifeboat
lb	pound
max	Maximum
MECO	Main Engine Cutoff
min	Minimum
[redacted]	[redacted]
N.M.	Nautical Mile

P.	Port
PDM	Pulse Duration Modulation
PMU	Pressure Make-Up Unit
[redacted]	[redacted]
PSAPO	Payload Sub Assembly Project Office
RF	Radio Frequency
rms	Root-Mean-Squared
rss	Root-Sum-Squared
S.	Starboard
[redacted]	[redacted]
sec.	Second
SL	Sea Level
SLC-1-E	Space Launch Complex - 1 - East Pad
SLC-3-W	Space Launch Complex - 3 - West Pad
SLP	Silicon Light Pulser
SLV	Standard Launch Vehicle
SRV	Satellite Recovery Vehicle
SS	Standard Stage
STC	Satellite Test Center
SV	Satellite Vehicle
TLM, T/M	Telemetry
TT&C	Telemetry, Tracking and Command
UDMH	Unsymmetrical Dimethylhydrazine
UHF	Ultra High Frequency
V	Vehicle Velocity
V/h	Ratio of Velocity to Altitude
VAFB	Vandenberg Air Force Base
VHF	Very High Frequency
[redacted]	[redacted]
w/o	Without
WTR	Western Test Range