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Vela Satellite Flight Planning

NOVEMBER 1966

Prepared by ALF. H. BOGEN
Systems Analysis Operations
Group II Programs
Satellite Systems Division
AEROSPACE CORPORATION

Prepared for COMMANDER, SPACE SYSTEMS DIVISION
AIR FORCE SYSTEMS COMMAND
LOS ANGELES AIR FORCE STATION
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El Segundo Technical Operations
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FOREWORD

This report is published by the Aerospace Corporation, El Segundo, California, under Air Force Contract No. AF 04(695)-1001. The report was authored by Alf H. Bogen of the Satellite Systems Division, in support of studies conducted for the Vela Program Office.

This report, which documents research carried out from March 1965 through August 1965, was submitted on 5 December 1966 to Col. Stephen H. Sherrill, Jr., SSUN, for review and approval.

Approved

W. L. Pritchard, Group Director
Group II Programs
Satellite Systems Division

Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

Col. Stephen H. Sherrill, Jr.
Space Systems Division
Air Force Systems Command

ABSTRACT

This report discusses the interrelation and operating aspects of the Vela satellite, the Titan IIIC launch vehicle and the ground support systems as they affect flight planning. Emphasis has been placed on the effects of constraints and conditions resulting from the mutual interaction of the systems.

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FIGURES

TABLES

CONTENTS

INTRODUCTION

General

The Vela Satellite Program is one of many approaches to implementation of a nuclear space weapons detection capability to monitor the Nuclear Test Ban Treaty. The program is sponsored by the Department of Defense (DOD) through the Advanced Research Projects Agency (ARPA), in conjunction with the Atomic Energy Commission (AEC). Technical management is assumed by the Space Systems Division (SSD) of the United States Air Force, with general systems engineering assigned to the Aerospace Corporation until the Fall of 1965.

Three pairs of Vela spacecraft have been successfully launched over a 3 year period, all of which are working at this time. Enormous quantities of data pertaining to space radiation and related phenomena have been obtained, and the result of this knowledge has had a profound influence on the spacecraft configuration contemplated for the next launch. This discussion describes the impact on flight planning brought about by the new highly modified Vela spacecraft and by assignment of the Titan IIIC (T-IIIC) booster system for the next launch. The following description of Vela system parameters is intended to provide sufficient background to acquaint the reader with previous launches.

The Present Vela System

The present Vela spacecraft consist of a space platform incorporating a complement of nuclear radiation sensors. These spacecraft are equipped with electronic gear to accommodate data storage, readout, command and communication capabilities. Electric power is provided by solar cells, and a rocket motor is furnished for injection into final orbit. The spacecraft vehicles are arranged in a tandem configuration as shown in Figure 1, facilitating launch of two spacecraft at a time. A spin-up mechanism, using a cold gas system, provides on-orbit inertial stabilization.

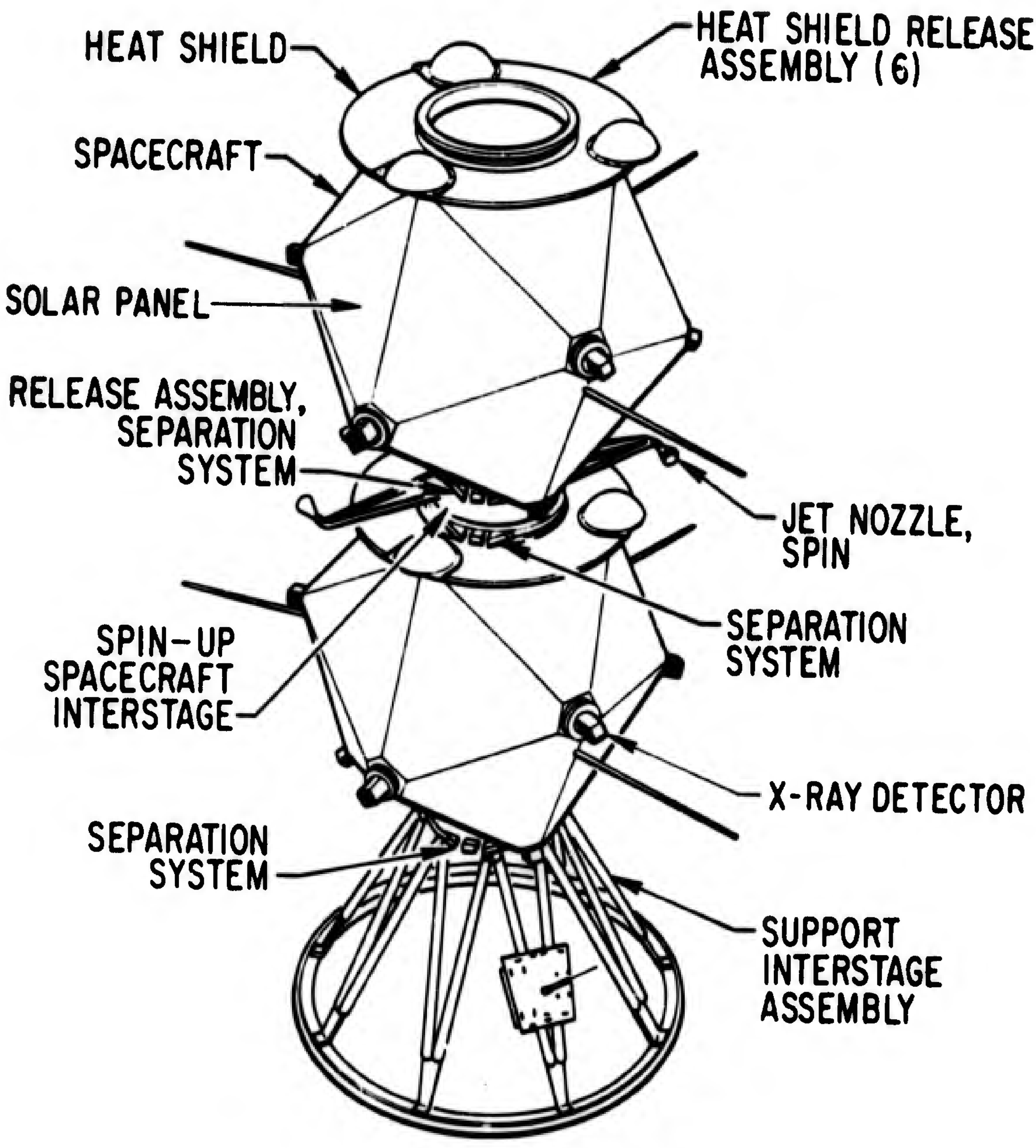


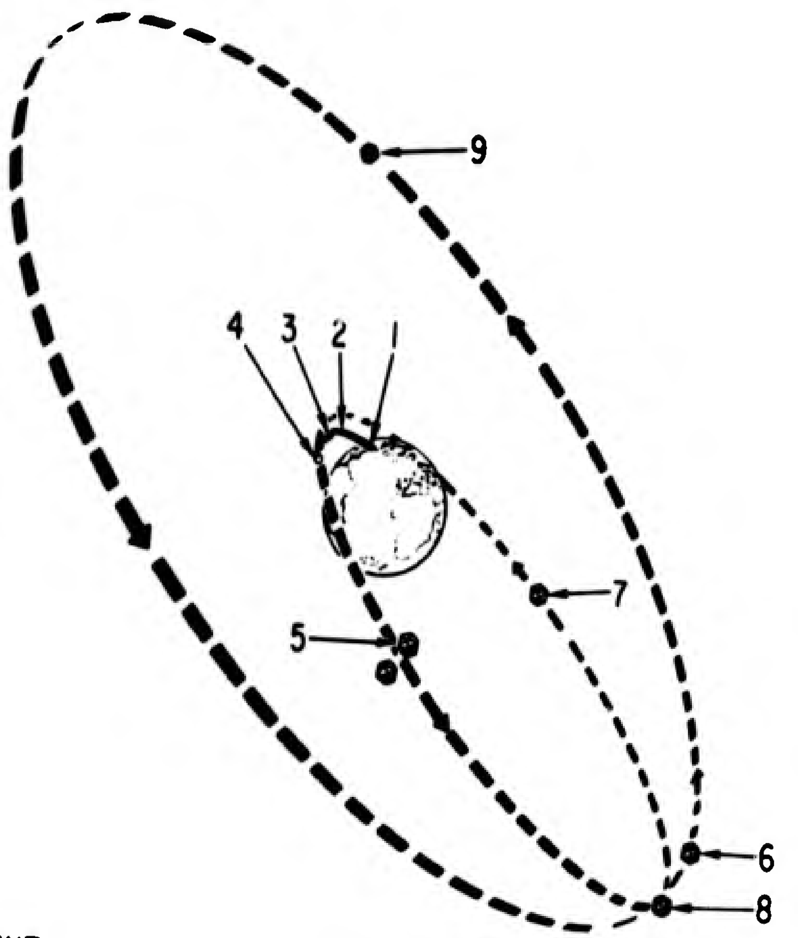
Figure 1. Vela Spacecraft System

On past missions the tandem spacecraft were boosted into a Hohmann transfer ellipse from Cape Kennedy, Florida, at a launch azimuth of 107 degrees by the Atlas/Agena booster combination. Following powered flight, terminated by the Agena engine burnout at a 106-nautical mile perigee altitude, the spacecraft were ejected from the booster, immediately spin-stabilized, and subsequently separated from each other. Following stabilization and separation the spacecraft coasted to an apogee of 55,000 nautical miles, where, at successive apogees, they were injected into final circular orbit. The injection sequence placed the spacecraft in the final orbit at an initial angular distance of approximately 140 degrees, as shown in Figure 2.

Velocity and position vectors subsequent to burnout were obtained by an on-board radar pulse beacon transponder and the TPQ-18 radars at the downrange Antiqua and Ascension Island tracking stations. Obtained radar data was immediately transmitted to the USAF Satellite Test Center in Sunnyvale, California, for generation of the ascent ephemerides by digital computing equipment. The acquisition, transmission, and processing of radar data at an early time, yielding fundamental information for establishing the spacecraft velocity and position vectors at a reference epoch near perigee, are very important. The knowledge of these vectors in turn permits definition of the time-position-velocity relationship at apogee, which enables control of the final orbit perigee altitudes as well as the closing and opening rate of the spacecraft orbit angular position after the spacecraft are injected into final orbit.

Transmission of the spacecraft rocket engine start signal at the proper time at successive apogees, providing necessary control of the desired final orbit characteristics, has been executed from the Vandenberg, Hawaii, and New Boston tracking stations, as well as from other stations. Obviously, tracking-station coverage at apogee and booster burnout is mandatory for the Vela operations.

In the Vela flight planning, another system parameter of importance has been the launch time and its related effects. This time is derived from the need to obtain the maximum angle between the spacecraft spin axis and the ecliptic



ITEM	EVENT
1	LAUNCH FROM ETR
2	STAGE I BURNOUT
3	STAGE II BURNOUT
4	SPACECRAFT SPINUP AND SEPARATION COMPLETE
5	SEPARATED SPIN-STABILIZED SPACECRAFT COAST TO APOGEE
6	FIRST SPACECRAFT INJECTION COMMAND (18 hr FROM LAUNCH)
7	SECOND SPACECRAFT CONTINUES IN TRANSFER ELLIPSE
8	SECOND SPACECRAFT INJECTION COMMAND (56 hr FROM LAUNCH)
9	FIRST SPACECRAFT POSITION AT SECOND SPACECRAFT INJECTION

Figure 2. Mission Profile

plane in order to obtain maximum solar-cell electric power output. This launch time also determines the spacecraft eclipse seasons. The spin-axis/ecliptic angle, initially defined by the spacecraft momentum vector at the time of spin-up, is adjustable by gradually torqueing the spacecraft, using a cold-gas system. In case of torqueing-system malfunction, this angle is unchangeable once established, and its maximum value is obtainable only when launch occurs at one discrete time each day, which changes at the rate of 24 hours per year. Thus, a time launch window was established based upon a tolerable minimum solar-cell power output. Fortunately, by appropriate selection of the day of launch, this launch time can be made compatible with other systems criteria, such as the effects on booster-attitude accuracy caused by Agena (Stage 2) horizon-sensor sun blinding and by spacecraft eclipse seasons resulting from a particular launch-time and date selection. Launch countdown procedures are also highly affected by these requirements. Figure 3 shows the factors affecting the launch schedule for previously conducted Vela space launches.

ADVANCED VELA FLIGHT SYSTEMS PARAMETERS

The overall Vela Satellite Program mission objectives for the next launch remain essentially unchanged as compared to previous launches. However, as previously noted, significant changes in systems requirements, such as the new heavier spacecraft configuration and use of the Titan IIIC booster system in lieu of the Atlas-Agena booster combination, have introduced far-reaching affects on launch and flight planning.

Spacecraft Systems Performance

The next Vela mission calls for a final circular orbit altitude of 60,000 nautical miles with an equatorial inclination of 33 degrees. This basically very simple requirement becomes complicated because of the transfer-orbit shaping constraints imposed by spacecraft performance and environmental criteria, and by booster performance, tracking, and mission operational considerations.

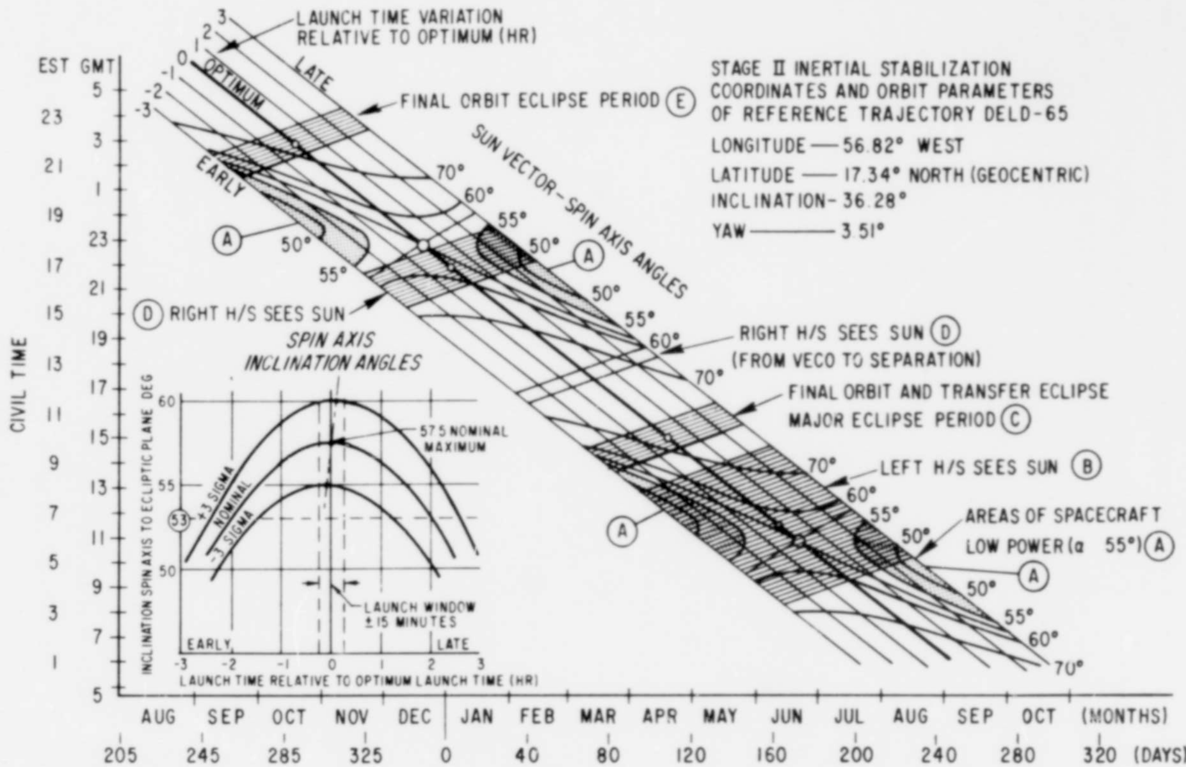


Figure 3. Factors Affecting Launch Schedule
(Optimum Launch Time)

The factor having the most important and influential affect on orbit shaping, considering the greatly increased spacecraft weight, was the decision to retain the same type and size of spacecraft rocket motor used in previous Vela spacecraft, but with a slight increase in total rocket impulse. This decision was made after considering the development costs associated with a new rocket motor having adequate impulse to permit use of the direct orbit (Hohmann) transfer method and was made possible only by the available payload capability provided by the Titan IIIC booster. The Vela spacecraft in its new and heavier configuration is capable of a final orbit injection velocity increment of 3,175 feet per second. This velocity increment is too small for circularization at 60,000 nautical miles using a Hohmann transfer ellipse perigee of 106 nautical miles. Therefore, the apogee velocity of the transfer orbit must be of such magnitude that when vectorially added to the spacecraft velocity increment capability the velocity sum defines the spacecraft velocity in the circular 60,000-nautical mile final orbit. An apogee velocity in excess of the apogee velocity of the direct transfer orbit (with a 106-nautical mile perigee altitude) can be regarded as an augmentation of the spacecraft rocket motor and automatically defines a final transfer orbit whose perigee altitude will satisfy the required injection conditions.

Booster Configuration

Launch will be conducted at Cape Kennedy, utilizing the Titan IIIC booster system and the integrated-test-launch (ITL) facility designed for this purpose. The flight azimuth of 107 degrees, which has been chosen, is compatible with an orbit equatorial inclination satisfying the Vela mission objectives and with the range safety requirements established by the National Range Division (NRD). The Titan IIIC booster system (which will be utilized by the Vela Satellite Program for the first time) consists of two solid strap-on motors having in excess of 1 million pounds of thrust each and three additional liquid-fueled stages. This booster system represents a formidable orbit payload capability. The booster final stage, or transtage, besides being used for orbit-injection purposes, is capable of intricate space maneuvers and contains the inertial-guidance, flight-control, and telemetry systems. The following sections outline the methods and conditions that have been used to satisfy the specified final orbit requirements when using this fixed booster configuration.

Orbit Transfer Method

The modified final transfer-orbit perigee altitude brought about by the impulse deficiency of the spacecraft rocket motor immediately introduces the following questions: (1) How are the spacecraft to be placed into this final transfer orbit; or, more specifically, what are the parameters of an additional or first transfer orbit and the transfer point which make such a transfer possible? (2) How does this transfer affect flight planning?

The basic constraint imposed upon transfer orbit shaping is the booster vehicle performance or, more precisely, the available booster velocity increment for the required total payload weight to be orbited. Therefore, before attempting to answer the first question it is desirable to evaluate the total orbit transfer velocity requirements and, in particular, to define an orbit transfer method which, if possible, results in a minimum transfer orbit velocity requirement and complies with the other pertinent requirements.

Figure 4 depicts a basic transfer method which features the use of bielliptic coplanar ellipses. Here the apsides of the first and final transfer ellipses are separated by an angle (θ) called true anomaly. The apogee altitude of the first ellipse is higher than the apogee altitude of the inside tangent ellipse by an overshoot altitude (ΔH). Booster velocity additions are envisioned to be applied at two locations; one at perigee of the first ellipse (ΔV_P) and the other at the first intersection or transfer point of the two ellipses (ΔV_F). The algebraic sum of these velocities is the total orbit velocity requirement ($\Sigma \Delta V$) demanded from the booster system. Specifically, this velocity requirement must be compatible with the booster available velocity for the total payload weight. It should be recognized that the above velocity sum is in addition to

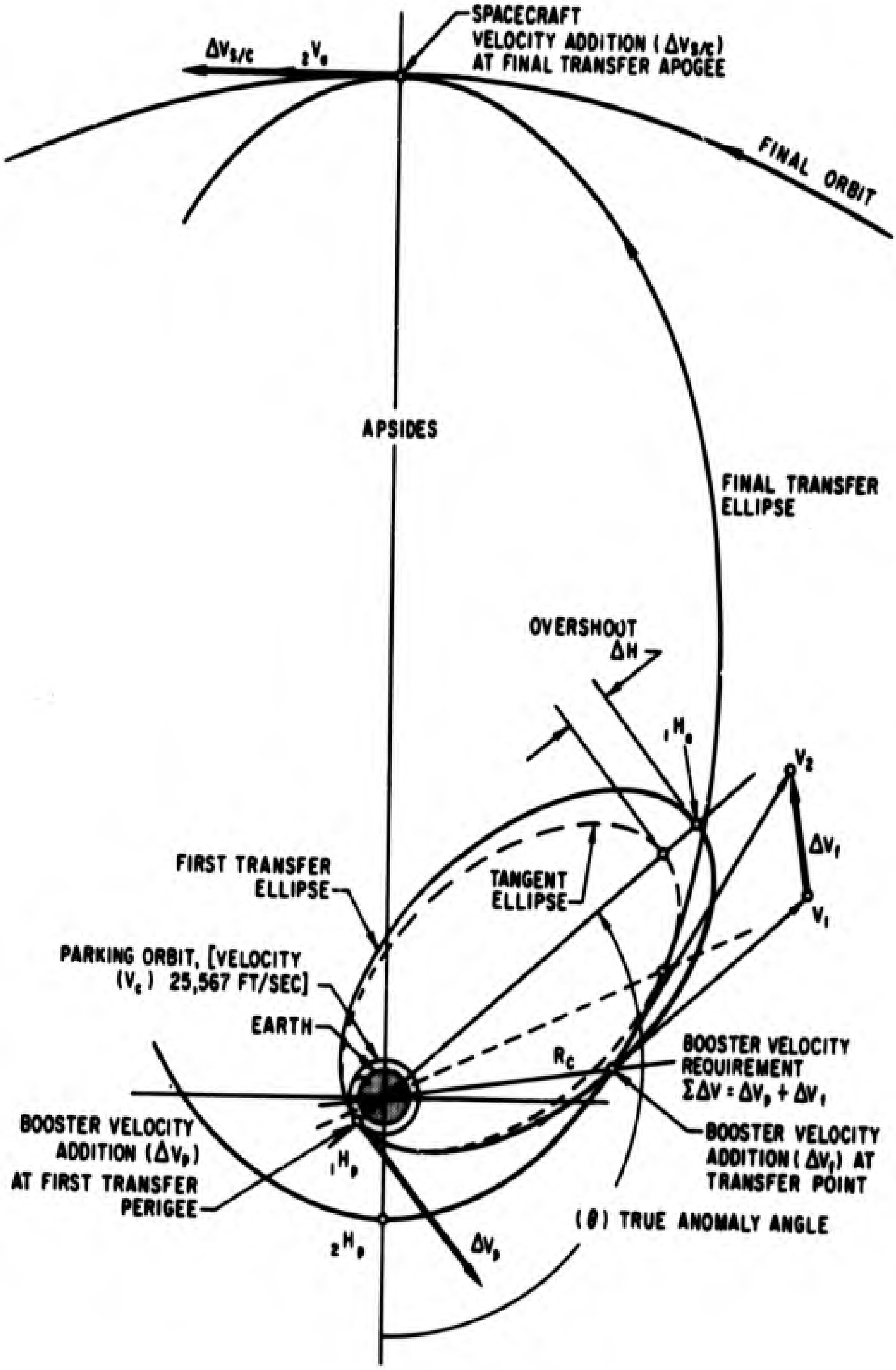


Figure 4. Flight Plan Orbit Definition

the reference velocity level represented by the circular velocity at the altitude of the first transfer ellipse perigee (V_C). By appropriate selection of the overshoot altitude (ΔH), this orbit-transfer scheme results in a minimum over-all energy or velocity condition when transferring into a final transfer orbit at a given transfer-point altitude. The final transfer-orbit apogee velocity for this transfer method is in excess of the corresponding Hohmann apogee velocity for equal transfer perigee altitudes. Although mathematical proof will not be developed here, this transfer method is analogous to the Hohmann transfer in the sense that under the above conditions it also represents a minimum energy expenditure. Although this is an extremely inefficient method of energy management, it is necessitated by the insufficient impulse of the spacecraft motor and is made possible by the large payload capability of the booster.

Implications of Flight Data Retrieval

The necessity to retrieve flight data information by telemetry for the purpose of evaluating various booster and spacecraft flight systems during the periods of transtage burns makes it mandatory to observe these events from some tracking station. In addition to knowing how well these systems perform, it also is essential to know where the vehicle is and where it is going. Information relating to position and velocity is vital for execution of proper orbit control. Specifically, it is necessary to establish the time at which the spacecraft final injections must occur in order to achieve desired control of the final orbit periods. These periods determine the spacecraft relative angular on-orbit position and ultimately the extent of global coverage. As a result, the final transfer-orbit parameters must be evaluated immediately subsequent to the transtage final burn (transfer point) based upon the instantaneous velocity and position vectors existing at this time. To do this the vehicle must be observed from some tracking station and the vector information retrieved. These vectors, derived from the transtage inertial guidance telemetry data and, redundantly, from the airborne radar pulse beacon transponder tracking information, largely eliminate the uncertainties of trajectory guidance dispersions and to some extent lunar and planetary perturbations. Usable retrieval of radar and telemetry data imposes additional flight planning requirements and indirectly influences orbit-shaping parameters. One of these is the on-board pulse beacon downlink signal strength, which imposes a limitation on the altitude of the transfer point in order to obtain unambiguous radar data. The tracking station selected to receive pulse beacon radar data must, besides being visible to the vehicle at the transfer point, be properly equipped and have a capability to immediately transmit this data to the central command and computing center for generation of the ascent ephemerides.

Orbit Shaping

A parametric evaluation of the pertinent total transfer orbit velocity requirement as a function of true anomaly at various overshoot altitudes, based on the orbit-transfer scheme discussed above, is displayed in Figure 5. This permits selection of the first transfer-ellipse parameters and definition of a transfer point to the last transfer ellipse which will comply with the major systems constraints of booster performance and radar pulse beacon (transfer point) range, and will at least give a preliminary answer to the question of how to accomplish a flight plan. Considering the over-all payload weight, it is seen from Figure 5 that the restrictions associated with the booster available velocity and the transfer-point altitude impose conditions of restricted opportunity for conducting the mission.

The extent of this opportunity is more clearly displayed in Figure 6 and is represented by the areas bounded by the regimes of excessive velocity requirements and transfer-point altitudes. By accepting higher altitudes and higher radar-signal attenuation levels, this opportunity is seen to improve rapidly. However, before a flight transfer point can be selected, the requirements of tracking-station visibility at the first perigee, the transfer point, and the first three final transfer-orbit apogees must be satisfied. Such a selection must also exhibit compatibility with spacecraft attitude criteria applicable

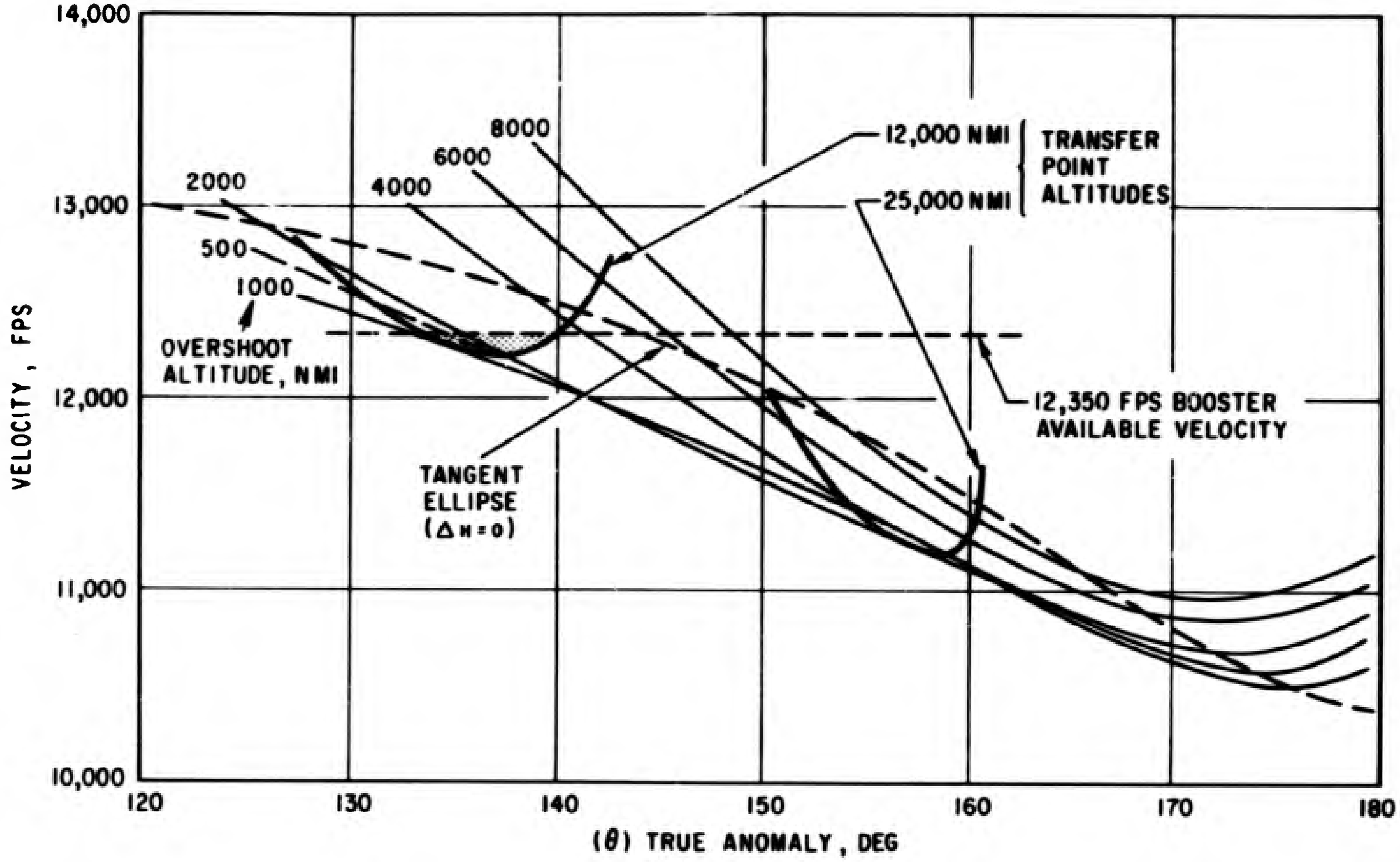


Figure 5. Orbit Velocity Requirement

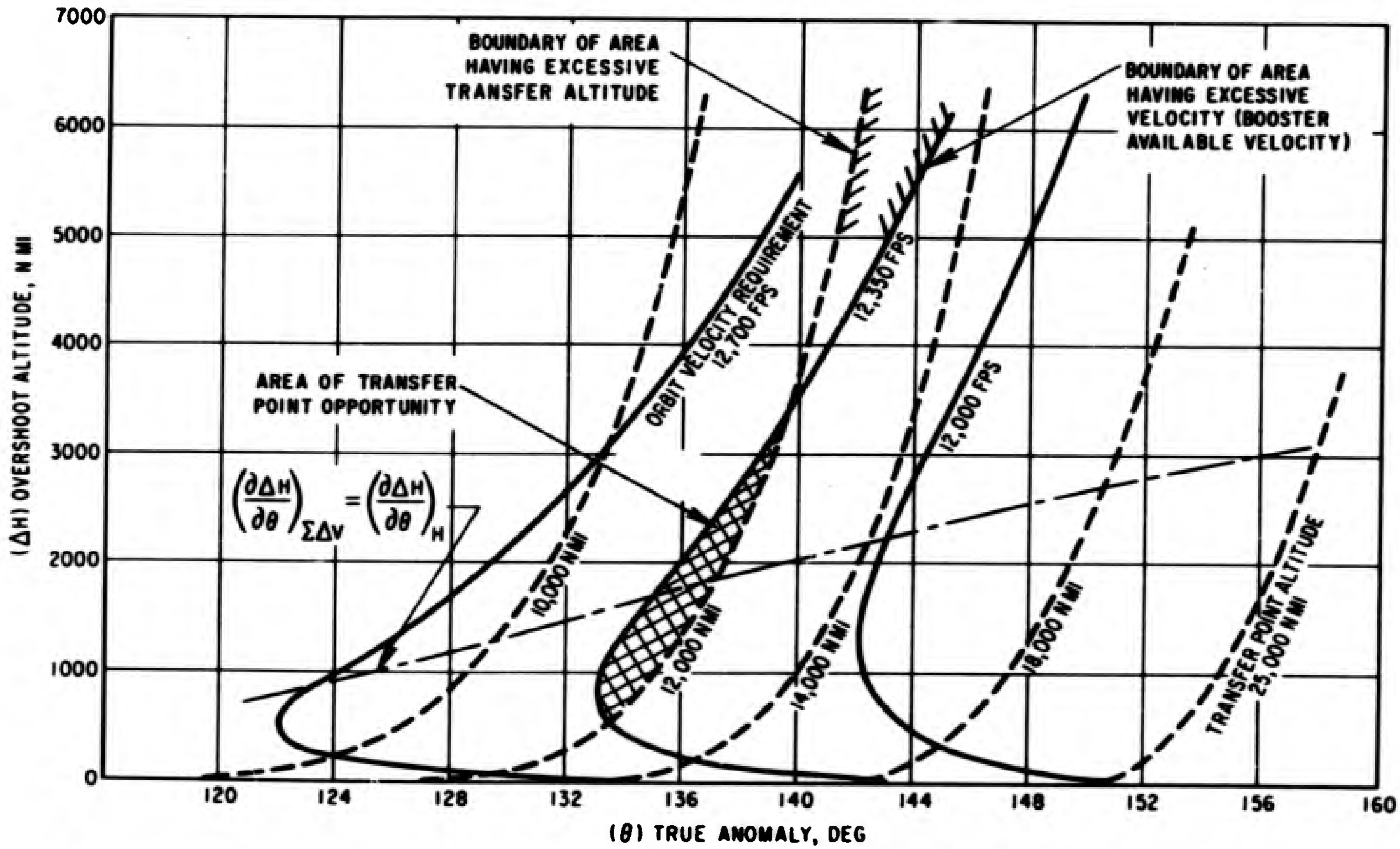


Figure 6. Transfer Point Constraints

to the transfer phase that originate from environmental and flight communication requirements. These criteria in turn affect basic reliability and orbit parameters, and require tracking stations to be immediately able to transmit the interrogated data to the central command and control center (near real-time transmission capability).

Tracking Station Visibility

The primary tool available to satisfy the station visibility requirement is utilization of a parking orbit at the altitude of the first transfer perigee. This in effect corresponds to an ambulating launch pad, the position of which is determined by the time spent in the parking orbit. An important factor in mission optimization is the trade-off between the selection of the low-altitude parking time and the time taken to reach the transfer-point altitude. Since the transfer-point altitude greatly influences the earth-vehicle relative position by virtue of its true anomaly angle relation, some increase in the time of arrival at a higher transfer point is introduced. Time-interval selection within this range is a consideration in the evaluation of over-all mission reliability because the transtage active flight system operating time, maneuvering complexity, and the number of transtage burns are involved.

Tracking-station visibility at the final transfer-orbit apogee is only slightly affected by the selection of the transfer-point altitude, since the total time taken for arrival at final apogee is relatively unchanged by variations in the transfer-point altitude. However, the final-orbit apogee visibility is greatly influenced by the low-orbit parking time.

Implications of Spacecraft Environment

The selection of parking time in conjunction with low or high transfer-point altitudes also is influenced by the previously mentioned criterion of spacecraft environment. This environment, determined by the spacecraft instantaneous thermal equilibrium conditions in the transfer orbits, is a function of equipment thermal characteristics and power levels, spacecraft external thermal radiation characteristics, and the direction by which the solar heat flux intercepts the spacecraft. With the possibility of three revolutions in the final transfer orbit, the spacecraft thermal conditions must yield acceptable minimum temperature levels for the solid rocket motor ignition conditions at consecutive apogees. These ignition temperature conditions demand that the sun vector be within 35 degrees of the perpendicular to the spin axis. Simultaneously, the spin axis must also be perpendicular to the major axis or line of apsides of the last transfer ellipse in order to accommodate proper velocity addition for injection into the final circular orbit, as indicated in Figure 7. Therefore, the spacecraft inertial-space attitude, directly derived from the above environmental and apogee orientation requirements, defines the launch time and, together with the basic equatorial inclination resulting from the chosen flight azimuth, also determines the orbit inclination relative to the ecliptic plane. The latter is instrumental in determining the duration of spacecraft eclipses in final orbit. In this connection, in order to insure a minimum duration of eclipses, it is desirable to obtain the highest possible ecliptic inclination (≥40 degrees). Therefore, the launch times associated with the compliance of the spacecraft spin-axis/sun-vector orientation and the orbit/ecliptic inclination requirements must be compatible and so selected that optimum conditions are attained. This launch time and the duration of any low-altitude parking also determine the occurrence and duration of spacecraft eclipses in the transfer orbits and must be considered in the flight planning because of spacecraft power demands and thermal requirements during this flight phase.

Orbit Transfer Sequence

Figure 7 shows the basic transfer scheme of the contemplated Vela mission. The transtage with the tandem spacecraft attached is horizontally injected at 100-nautical miles altitude. Subsequent to this injection, occurring at the perigee of the first transfer ellipse, the vehicle coasts to the transfer-point

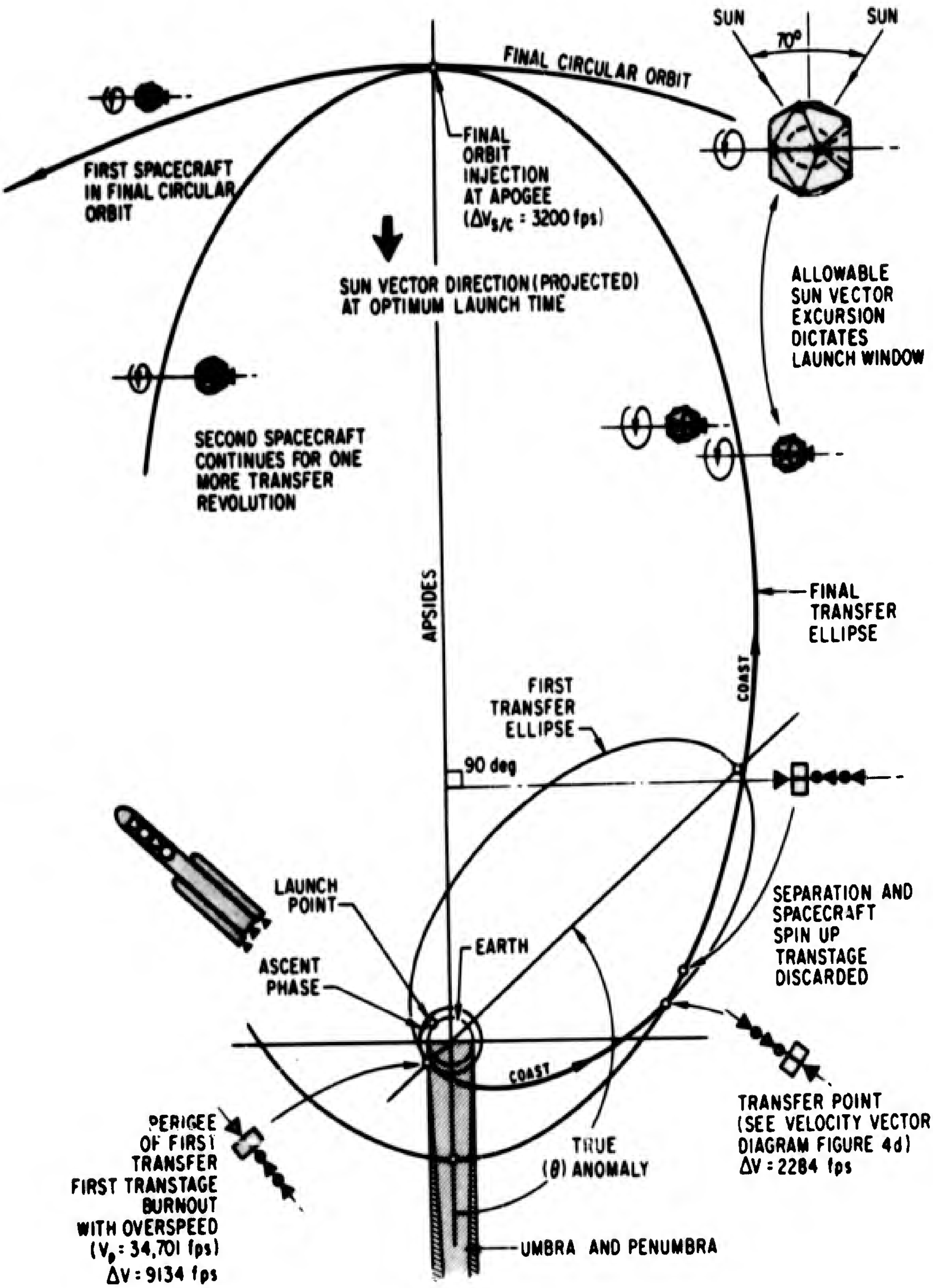


Figure 7. Basic Transfer Sequence and Attitude Orientation

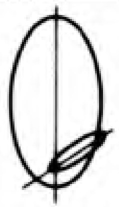
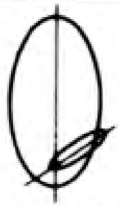
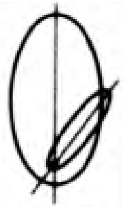
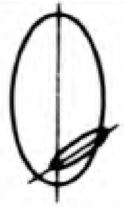
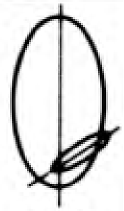
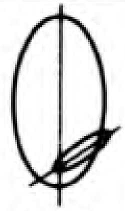
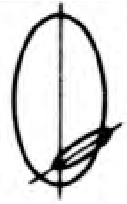
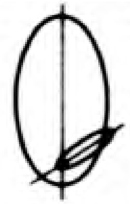
altitude. During this coast period the vehicle attitude is oriented in such a manner that the sun vector is perpendicular to the vehicle longitudinal axis. Short-duration excursions in vehicle attitude are made to satisfy vehicle communication-antenna pattern (telemetry) requirements. Shortly before arrival at the transfer point, the transtage is so oriented that the final burn will result in the proper velocity vector magnitude to achieve the desired characteristics of the last transfer ellipse. Following the transfer-point burn, the vehicle is oriented to permit interrogation of the radar beacon transponder for velocity and position vector determination, and is subsequently reoriented in the orbit plane with the vehicle center line perpendicular to the line of apsides of the last transfer ellipse. When the proper attitude is attained the tandem satellite pair is ejected from the transtage, immediately spin stabilized, and subsequently separated. At this point the transtage is backed away from the spacecraft to preclude any possible collision and its functions terminated. The two spinning spacecraft, now in final transfer orbit, continue to coast until they are injected into the final circular orbit by ground command at successive apogees at a time determined by the final transfer-orbit ephemerides.

VELA FLIGHT PLAN SELECTION

Many possible flight plans exist employing the orbit-transfer method discussed above. The ultimate choice of plan is governed by the probability of over-all mission success, the extent and quality of data retrieved for the purpose of evaluating flight systems performance, and considerations of systems and operational costs.

Table 1 displays some flight plans considered in the selective process of obtaining a technically feasible approach. The four major technical criteria of tracking-station visibility, transfer-point and separation altitude, time to reach these altitudes, and eclipse duration with and without low-altitude orbit parking are compared for each plan.

Table I. Summary of Investigated Flight Plans

	FLIGHT PLAN	1	2	3	4	5	6	7	8
									
	CRITERION	$\Phi = 135$ 2 BURN $t_c = 0$ MIN	$\Phi = 135$ 2 BURN $t_c = 0$ MIN	$\Phi = 155$ 2 BURN $t_c = 0$ MIN	$\Phi = 135$ 3 BURN $t_c = 11$ MIN	$\Phi = 135$ 3 BURN $t_c = 22$ MIN	$\Phi = 135$ 3 BURN $t_c = 33$ MIN	$\Phi = 135$ 3 BURN $t_c = 40$ MIN	$\Phi = 135$ 3 BURN $t_c = 90$ MIN
1	VISIBILITY AT 100 N MI BURNOUT	DOWNRANGE SHIP	DOWNRANGE SHIP	DOWNRANGE SHIP	ANTIGUA	ANTIGUA	ANTIGUA	ANTIGUA	ANTIGUA
2	VISIBILITY AT PERIGEE	DOWNRANGE SHIP	DOWNRANGE SHIP	DOWNRANGE SHIP	ASCENSION	PRETORIA	INDIAN OCEAN SHIP	CARNARVON	ANTIGUA
3	VISIBILITY AT LAST BURN	CARNARVON PRETORIA	CARNARVON	ASCENSION	PRETORIA CARNARVON	HULA	BOSS COOK HULA	COOK	ASCENSION
4	VISIBILITY AT SEPARATION	CARNARVON PRETORIA	ASCENSION	ASCENSION	PRETORIA CARNARVON	HULA	BOSS COOK HULA	COOK	ASCENSION
5	VISIBILITY AT S/C INJECTION (APOGEE)	GOOD	GOOD	GOOD	GOOD	GOOD	ACCEPTABLE	BAD	ACCEPTABLE
6	TRANSFER POINT ALTITUDE	12.000	25,000	25.000	12,000	12,000	12,000	12,000	12,000
7	MISSION TIME (MIN)		240	240	131	142	153	160	205
8	ECLIPSE DURATION (MIN)	30	30	< 30	30	30	30	30	60
9	CAUSE OF FLIGHT PLAN REJECTION	NEAR REAL TIME DATA TRANSMISSION (3,4)	NEAR REAL TIME DATA TRANSMISSION (3)		NEAR REAL TIME DATA TRANSMISSION (3,4)	TRACKING STATIONS EQUIPMENT (2)	TELEMETRY SHIP LOGISTIC PROBLEM (2)	APOGEE VISIBILITY (5)	ECLIPSE DURATION (8)

The visibilities listed at each tracking station are based upon a five-degree horizon at each of the pertinent points of the transfer orbits. In addition, the visibility at the last transfer apogee is termed good, acceptable, or bad, depending upon the duration and continuity of how the spacecraft are visible to the tracking stations within a four-hour time interval on either side of the time of reaching nominal final transfer-orbit apogee. The transfer-point and/or spacecraft-separation altitude is the actual altitude in nautical miles, limited either by booster performance (low) or by the condition where a minimum change in true anomaly angle makes it visible to the station (high). The eclipse duration is the total time the spacecraft spends in the combined umbra and penumbra before reaching apogee. With the information presented in Table 1, it is possible by the process of elimination to make a technical judgment as to which approach is most meritorious and to provide an answer to the original question of how to define a satisfactory flight plan.

The flight plans fall into the two main categories of (a) employing a parking orbit ($t_c > 0$) and (b) not employing a parking orbit ($t_c = 0$). Table 1 presents two flight plans, one in each category (No. 3 and No. 6), which are potentially acceptable plans. However, these candidate plans are associated with logistic problems, since they require deployment of tracking ships to the first-transfer perigee locations. It is apparent that Flight Plan No. 6 requires a tracking ship in the Indian Ocean area and that Flight Plan No. 3 requires a tracking ship between the Ascension and Antigua Islands. Flight Plan No. 3 does have the disadvantage of a high transfer-point altitude, but the location of the tracking ship is only 1500 nautical miles from the home port as compared to 12,000 nautical miles for Flight Plan No. 6.

The reason for rejecting other flight plans is based upon noncompliance with mandatory requirements relating to the payload, ground, and launch systems, a situation which cannot be easily improved within the time period allotted.

The ultimate choice therefore simplifies itself to the relative disadvantage of accepting a higher transfer-point altitude with attendant higher radar-signal attenuation levels, or dispatching a tracking ship to the Indian Ocean area.

The selected Flight Plan No. 3 is not ideal, but may be considered the best plan in view of the alternatives. The chosen flight plan satisfies the requirement of launch time compatibility as shown in Figure 8, where the launch opportunity or launch window for the contemplated launch date accommodates compliance with the spacecraft environmental criteria as well as with the high orbit ecliptic inclination required for minimum duration of final-orbit eclipses. Figure 9 shows the transfer-orbit ground track and Figure 10 the visibility available at the three consecutive transfer apogees. Table 2 lists the more important transfer-orbit characteristics for this flight plan.

Because of the relative complexity involved in the areas of transtage guidance and attitude orientation control, as well as in the over-all operations of the spacecraft, the selected flight plan may well represent one of the more ambitious space ventures in which the Titan IIIC booster system will participate.

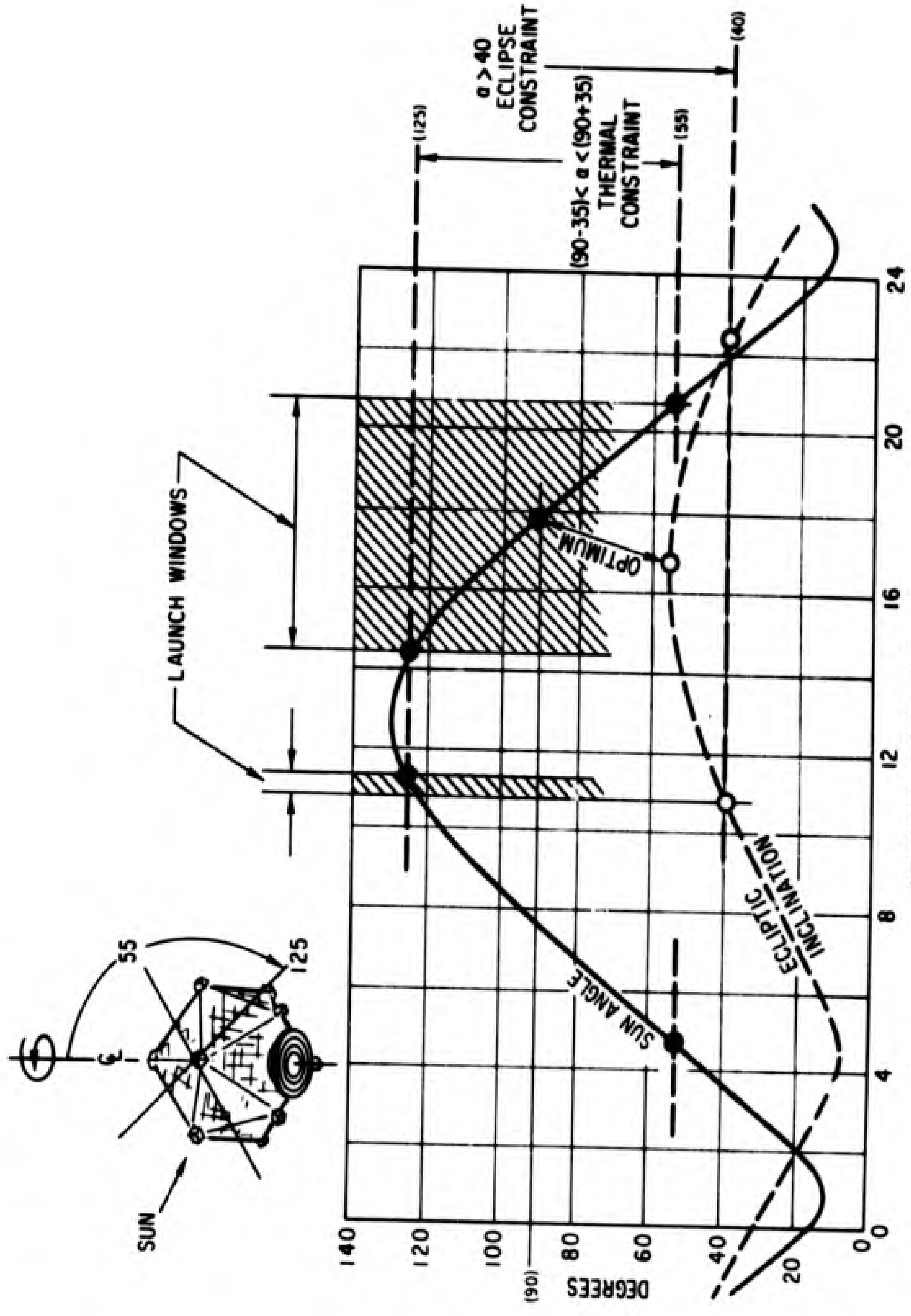


Figure 8. Launch Window

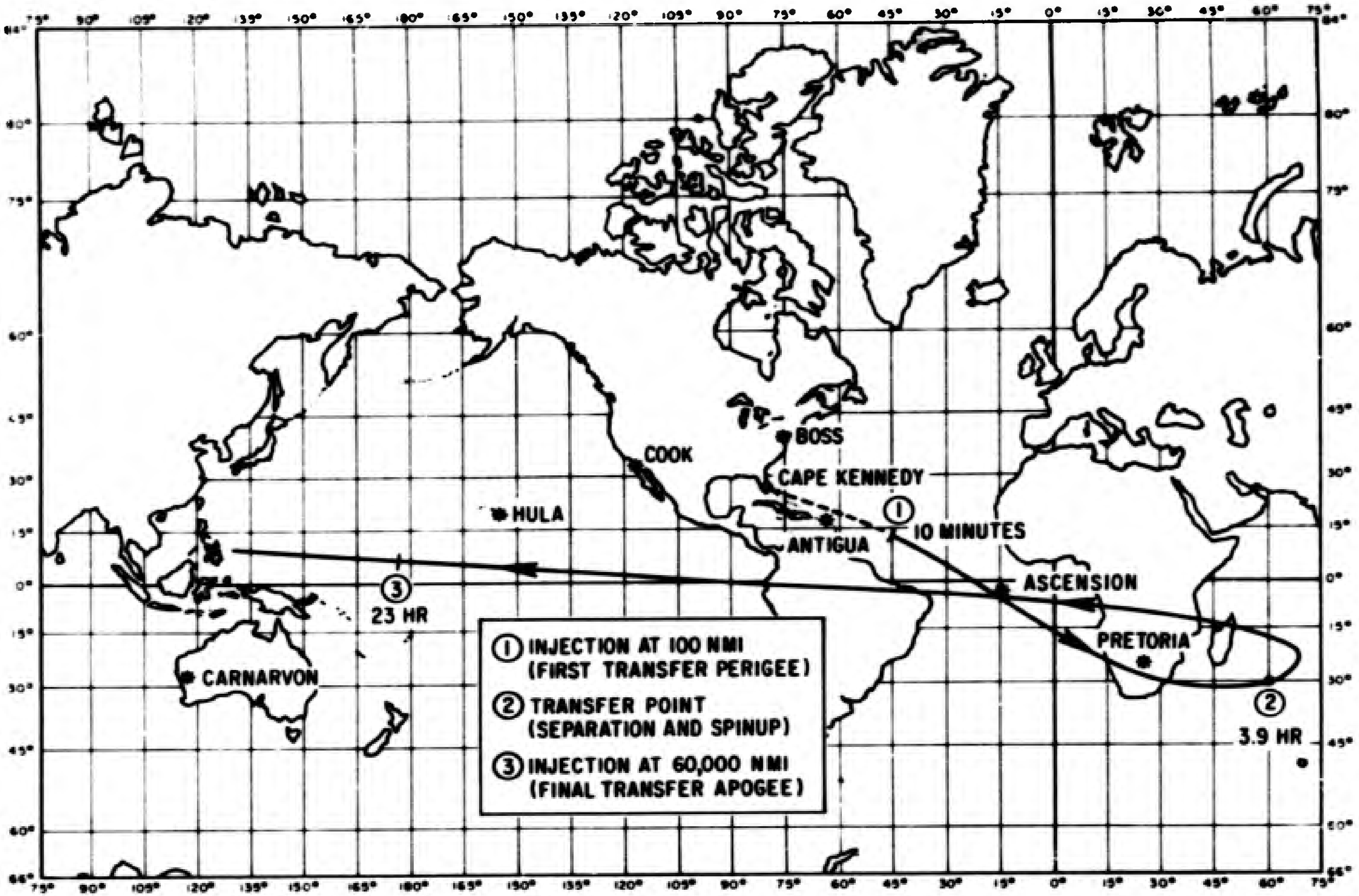


Figure 9. Ground Trace

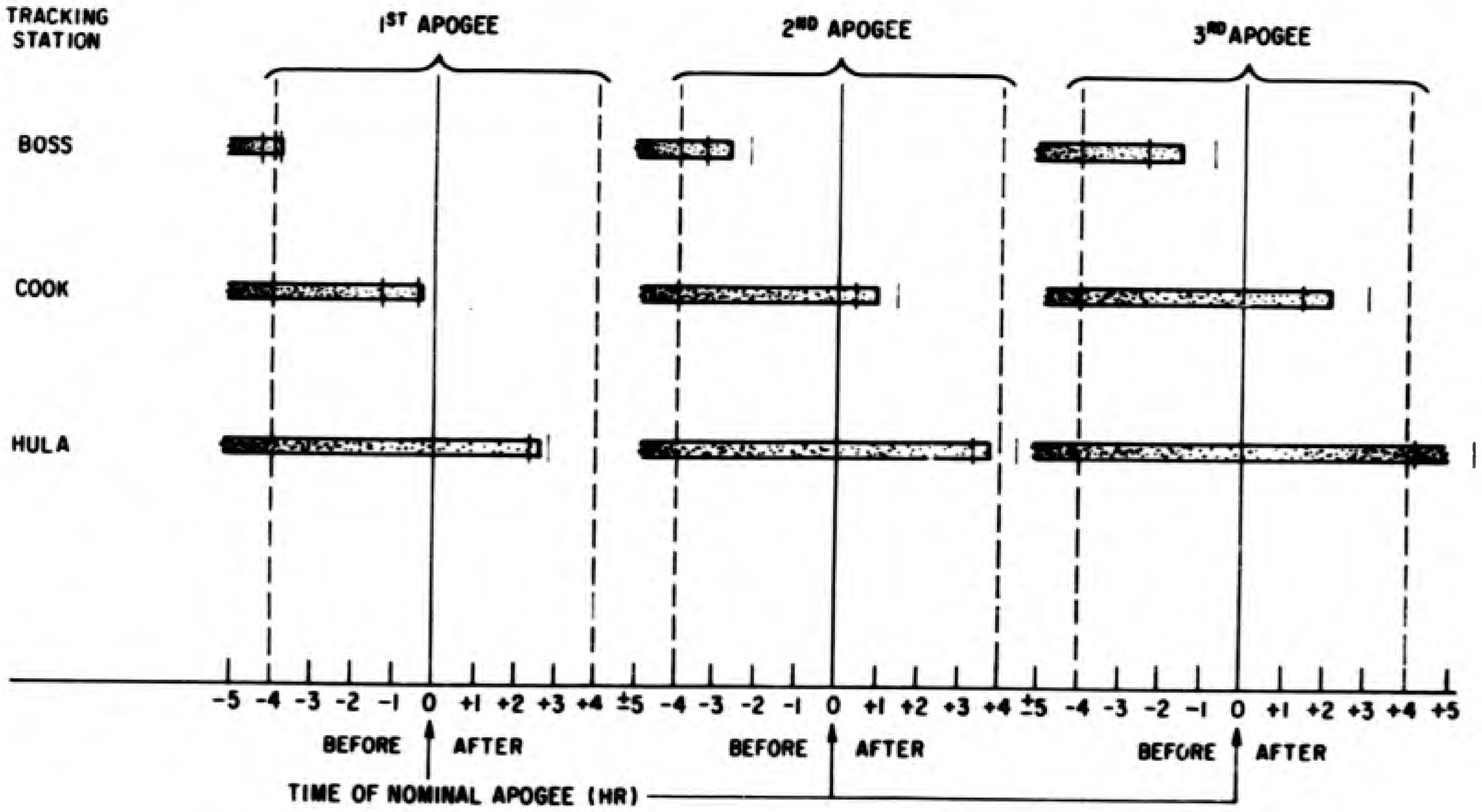


Figure 10. Apogee Visibility

Table II. Vela/Titan IIIC Orbit Parameters (Approximate Values)

• FINAL CIRCULAR ORBIT ALTITUDE	60,000 N MI
• FIRST TRANSFER ELLIPSE APOGEE	37,824 N MI
• OVERSHOOT ALTITUDE (ΔH)	1200 N MI
• TRANSTAGE FIRING ALTITUDE	24,000 N MI
• ELAPSED TIME BEFORE TRANSTAGE LAST BURN	3 HRS 51 MIN
• FIRST TRANSFER ELLIPSE PERIGEE	103 N MI
• TRUE ANOMALY ANGLE (θ)	155°
• SECOND TRANSFER ELLIPSE APOGEE	60,000 N MI
• SECOND TRANSFER ELLIPSE PERIGEE	4,611 N MI
• PERIOD OF SECOND TRANSFER	47 HRS
• PERIOD OF FINAL ORBIT	111.3 HRS
• PERIOD OF FIRST TRANSFER	23.4 HRS
• INITIAL ORBIT ANGULAR SPACING	152°
• FINAL ORBIT ANGULAR SPACING	180°

UNCLASSIFIED

Security Classification

DOCUMENT CONTROL DATA - R&D

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13 ABSTRACT

This report discusses the interrelation and operating aspects of the Vela satellite, the Titan IIIC launch vehicle and the ground support systems as they affect flight planning. Emphasis has been placed on the effects of constraints and conditions resulting from the mutual interaction of the systems.

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14

KEY WORDS

Bielliptic Coplanar Ellipses
Final Orbit Perigee
Final Orbit Periods
Flight Plans
Hohmann Transfer
Inertial-Space Attitude
Injection Velocity
System Constraints
Minimum Energy Expenditure
Mission Reliability
Orbit Transfer
Parking Orbit
Separation Altitude
Solar Heat Flux
Spacecraft Eclipses
Spacecraft Momentum Vector
Time Launch Window
Time to Reach Altitudes
Titan IIIC Booster System
Tracking Station Visibility
Transfer Point Altitude
Transfer-Orbit Shaping Constraints
Vela Spacecraft
Velocity and Position Vectors

Abstract (Continued)

SUPPLEMENTARY

INFORMATION

AEROSPACE CORPORATION

Post Office Box 95085, Los Angeles, California 90045, Telephone 648-5000

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1 February 1967

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To: Copyholders

The subject report was inadvertently published without a statement of acknowledgment, which should have appeared on page iv. The author wishes to take this means of extending his appreciation of assistance to all who directly or indirectly contributed to the formulation of the Vela Satellite Flight Planning.

Especially, this acknowledgment is extended to Mr. Andrew H. Milstead, formerly of the Astrodynamics Department, Electronics Division, and Mr. Hans E. Hogfors, Guidance Dynamics Department, Electronics Division, both of the Aerospace Corporation, who generated and documented orbit mechanics data based upon computer programs specially designed by them for the Vela program. In addition, this acknowledgment is also extended to Mr. Charles A. Noel, Space Launching Systems Development Directorate, Manned Systems Division, and Mr. Robert A. Rapuano, Vela Program Office, Satellite Systems Division, of the Aerospace Corporation, and Mr. Lee Ornella of TRW Systems, whose input and invaluable advice made this report possible.

It is hoped that this letter will be attached to the subject report (as page iv) and serve as grateful acknowledgment to the above.

Sincerely,

Alf H. Bogen

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