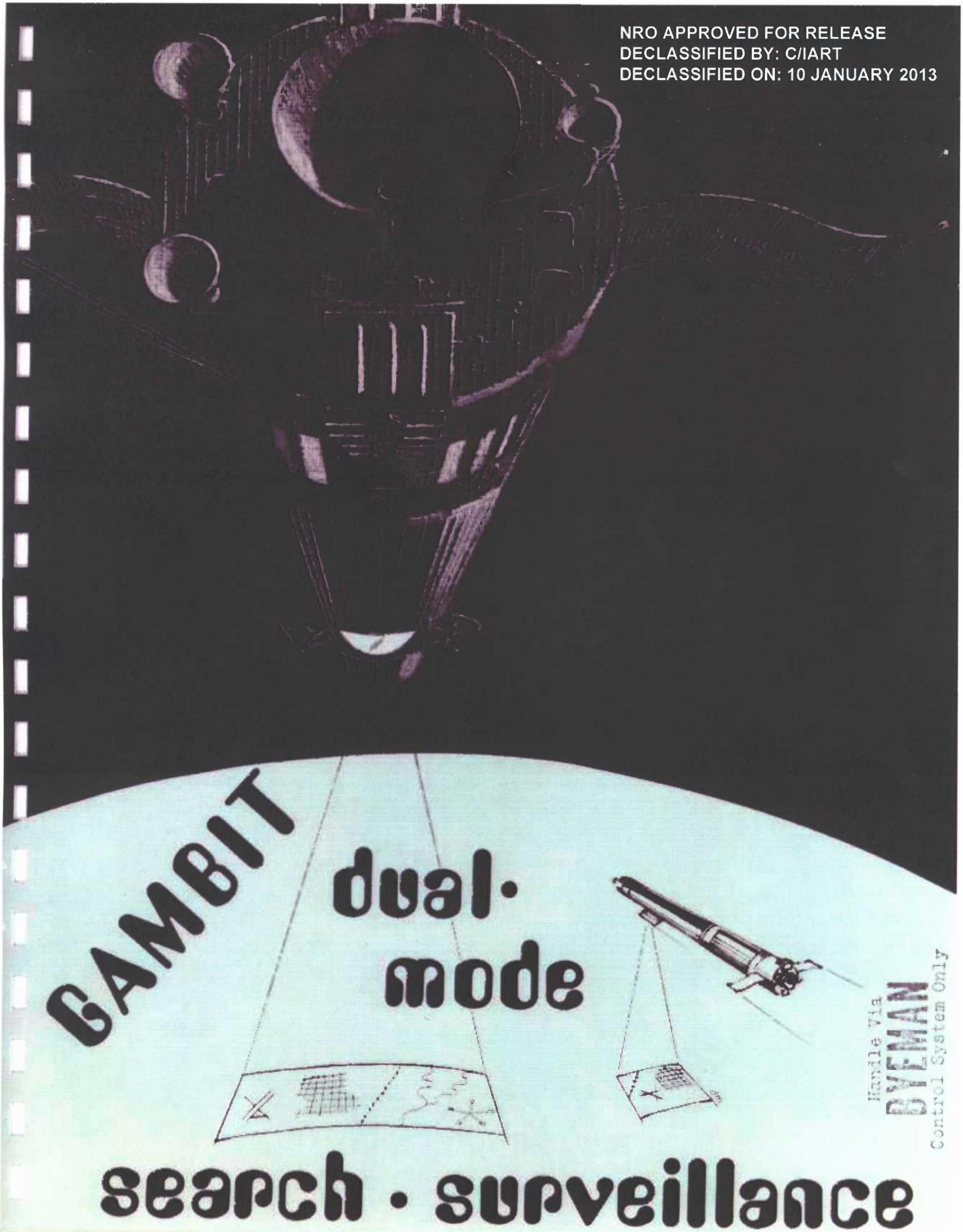
GAMBIT
dual-
mode
search · surveillance
Handle Via
BYEMAN
Control System Only

FOREWARD

In response to a need established by the expanding role of the GAMBIT vehicle, this Dual Mode Orbital Planning Document has been prepared by the Program Office to provide mission planners an understanding of the full capabilities of the hardware and the orbital options available. The multitude of planning variables prohibits identification of the exact capabilities of a specific vehicle, however, the data presented is a good estimate of the capabilities of the GAMBIT Dual Mode Block of vehicles (i.e., Vehicles 50-54 when modified to the Dual Mode configuration).

Acknowledgement is made of the support provided by [redacted] and [redacted] whose technical expertise contributed greatly to the preparation of this report.

[redacted]
Program Office

Controlled

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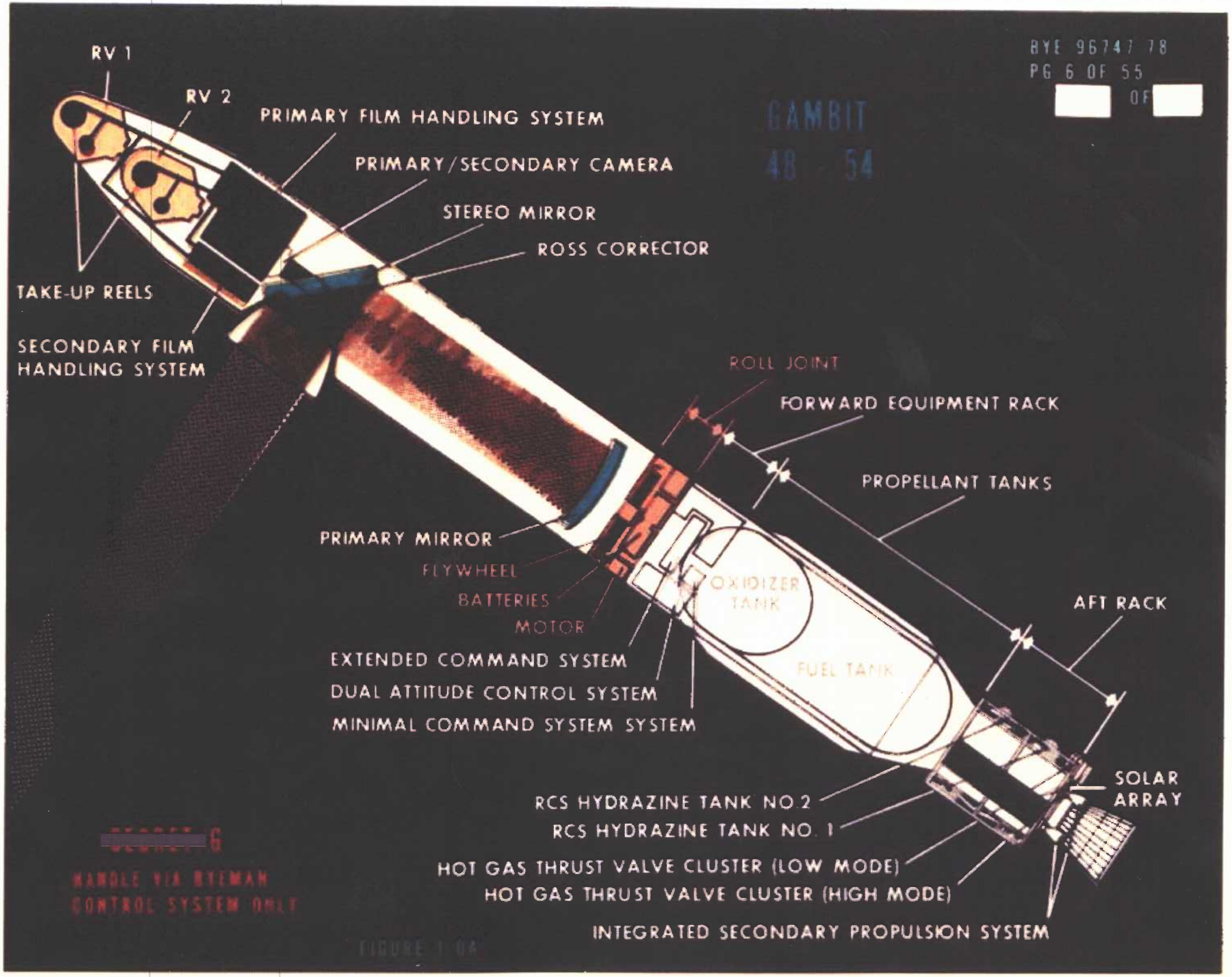
LIST OF FIGURES

THE DUAL MODE MISSION PLAN

1.0 THE MISSION OBJECTIVE - The GAMBIT mission has been redefined as an on-call backup photographic resource capable of responding quickly to a need for photography that is not being supplied by other systems. The current satellite vehicle design is being modified to provide a Dual Mode capability; that is, the flexibility of performing a low altitude high resolution surveillance mission and/or a high altitude wide area search mission (or any altitude in between). When modifications are complete, a Dual Mode vehicle will be maintained in a continuous 45-day readiness state.

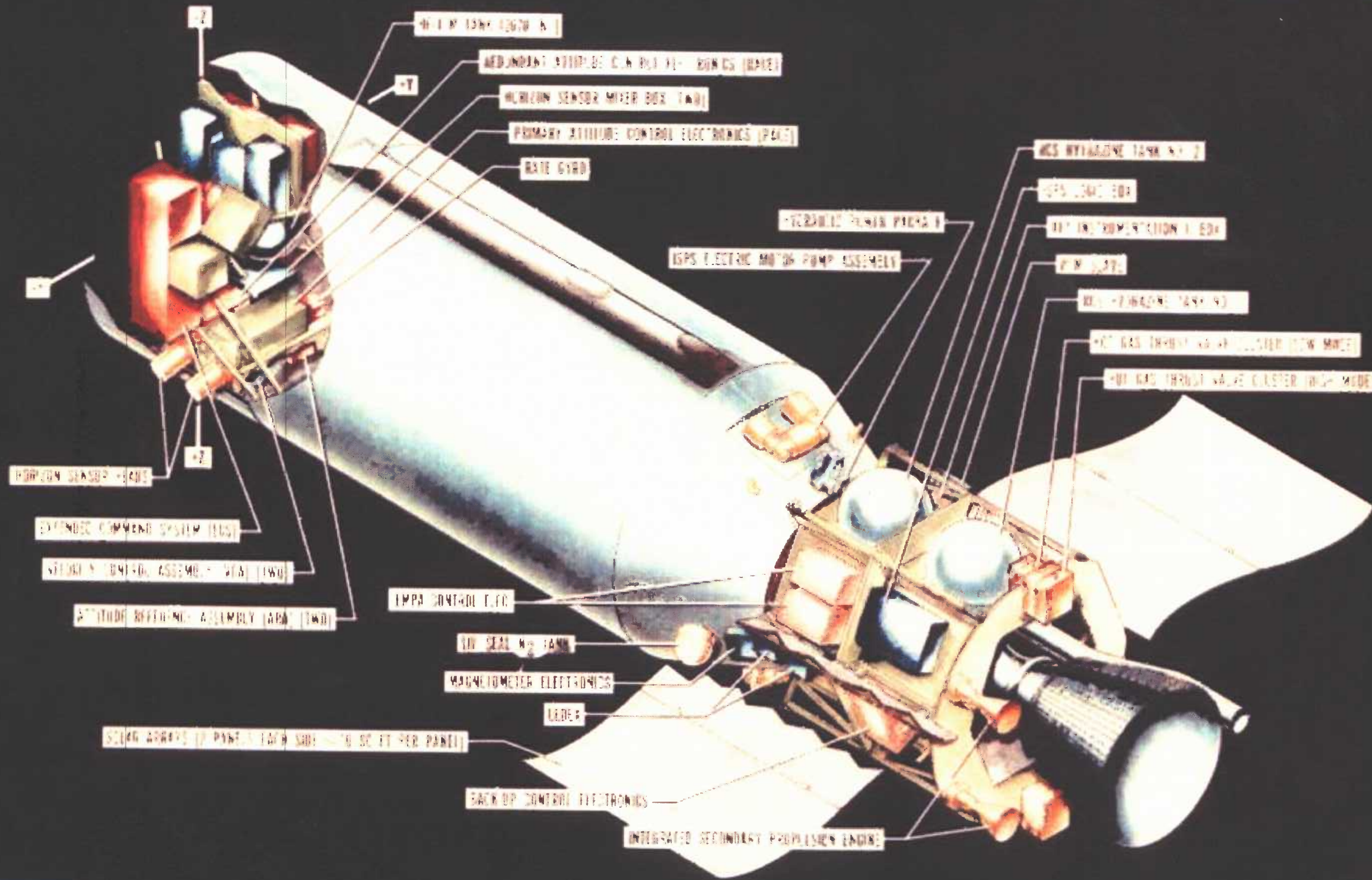
The primary objective of the Dual Mode Program is to develop and maintain the capability to launch and perform photographic missions within 45 days of a routine call-up. This can only be accomplished by having a majority of the pre-flight planning requirements fully developed and continuously maintained in a quick-reaction readiness state. The purpose of this report is to document the Dual Mode orbit planning portion of the pre-flight requirements.

This report contains the background data and analysis used in the study of possible Dual Mode mission scenarios. Every emphasis in this study has been directed at utilizing the full capability of the hardware with the realization that the specific requirements of the call-up mission may impose additional limitations.


BYE 96747 78
PG 6 OF 55
OF
GAMBIT
48 - 54
RV 1
RV 2
PRIMARY FILM HANDLING SYSTEM
PRIMARY/SECONDARY CAMERA
STEREO MIRROR
ROSS CORRECTOR
TAKE-UP REELS
SECONDARY FILM HANDLING SYSTEM
ROLL JOINT
FORWARD EQUIPMENT RACK
PROPELLANT TANKS
PRIMARY MIRROR
FLYWHEEL
BATTERIES
MOTOR
OXIDIZER TANK
FUEL TANK
AFT RACK
EXTENDED COMMAND SYSTEM
DUAL ATTITUDE CONTROL SYSTEM
MINIMAL COMMAND SYSTEM SYSTEM
SOLAR ARRAY
RCS HYDRAZINE TANK NO.2
RCS HYDRAZINE TANK NO. 1
HOT GAS THRUST VALVE CLUSTER (LOW MODE)
HOT GAS THRUST VALVE CLUSTER (HIGH MODE)
INTEGRATED SECONDARY PROPULSION SYSTEM
~~SECRET~~-G
HANDLE VIA BYEMAN
CONTROL SYSTEM ONLY
FIGURE 1-04

BYE 96747-78
PG 7 OF 55
CY [redacted] OF [redacted]

VEHICLE 48 SATELLITE CONTROL SECTION



[redacted]

[redacted]

FIGURE 1.0 8

SECRET/G

HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

DUAL MODE ORBIT PLANNING CONSTRAINING PARAMETERS

2.0 CONSTRAINING PARAMETERS - The parameters which strongly influence the development of mission scenarios are shown in Figure 2.0. Normally the mission requirements are defined in the intelligence community well in advance of the planned launch date. The Program Office then uses the mission definition to develop the orbit plan which will utilize all vehicle resources to satisfy or exceed these requirements while observing all hardware constraints. However, the new quick-reaction concept requires a different approach. The lack of a defined launch date, the expanded orbit options available, and the desire to minimize the timeline of activities between call-up and launch make it imperative that the entire community be aware of the full range of GAMBIT Dual Mode hardware capabilities. Therefore, this report assumes no specific mission definition. Instead, it identifies the full range of orbit options the Dual Mode vehicle can support for use by the community in the development of GAMBIT mission requirements.

2.1 PARAMETERS AFFECTING MISSION DURATION - The most critical factor influencing mission duration is the amount of propellants remaining after injection into orbit. A nominal prelaunch propellant load consists of approximately 14,000 lbm total of Unsymmetrical Di-methal Hydrazine (UDMH - fuel) and Fuming Nitric Acid (HDA - oxidizer). These propellants are used by the Main Engine (ME) for orbit injection and by the Integrated Secondary Propulsion System (ISPS) for daily drag make-up, ground trace spacing control, transfer orbits, emergency orbit adjusts, and deboost. Several orbit planning parameters directly influence how much of this load is required for orbit injection and, therefore, how much will remain for on-orbit use. These include:

Battery complement	Injection inclination	Injection altitude profile
Solar array power input	Injection argument of perigee	Argument of perigee control

NOTE: Mission definition remains the primary influencing factor, however, this report assumes no specific mission requirement.

Approximately 2,000 lbm of propellants will remain after injection. This could vary by several hundred lbm depending upon booster performance and the selection of the above parameters.

DUAL MODE ORBIT PLANNING
CONSTRAINING PARAMETERS

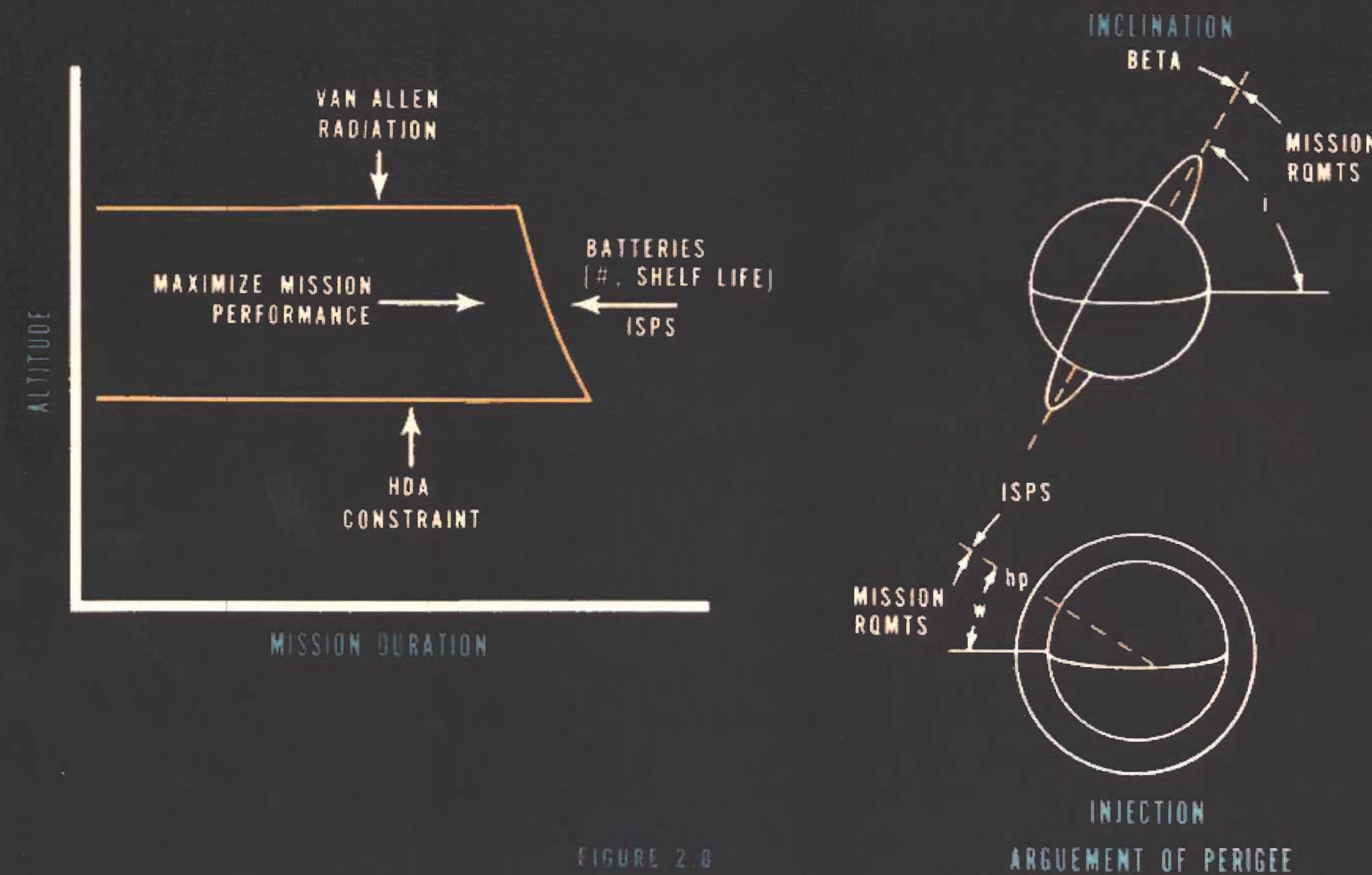


FIGURE 2.0

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HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

PARAMETERS AFFECTING MISSION DURATION

2.1.1 POWER REQUIREMENTS - The number of batteries needed to support the power requirements for a given mission is dictated by the mission duration, altitude profile, heater requirements, solar array efficiency, subsystem use rates, sun angle (beta), etc. The GAMBIT vehicle has the ability to carry eight (8) batteries; however, existing data indicates missions of up to 120 days at a maximum altitude of 450 nautical miles can be accomplished with a three (3) to five (5) battery configuration.

The selection of the specific battery configuration is the result of trade-off studies involving the mission requirements and the above parameters. The impact of adding one battery to the vehicle baseline configuration (4 batteries) is to increase the vehicle dry weight by 166 lbm. This is convertible to a reduction of approximately 145 lbm of on-orbit ISPS propellants. In terms of orbital life, 145 lbm of ISPS propellants equates to approximately seven (7) days of mission life in a 75 NM by 200 NM orbit.

2.1.2 SUN ANGLE (BETA) - The spacecraft thermal or power consideration require that the angle between the sun and the orbit plane be maintained within specific bounds for the duration of the mission. This angle, called the beta angle, is a function of solar declination, orbit inclination, and the difference in right ascensions of the true sun and the ascending node. The first quantity depends upon launch date, the second is essentially constant during the mission, and the third is influenced by the nodal regression and seasonal variation.

The variability of these quantities prevents the beta angle from being held perfectly constant throughout a mission. However, it is possible to select conditions which will maintain the beta angle within the prescribed tolerable range of values. The two conditions most accessible to mission planners are inclination and launch window.

The effect of inclination on beta angle for a specific orbit profile can be seen in Figure 2.1.2 A. The sun-sync inclination for the orbit shown (75 NM by 200 NM) yields a near constant mission length beta. However, as inclination deviates from this, the beta angle varies more widely.

BETA ANGLE VARIATION VS INCLINATION

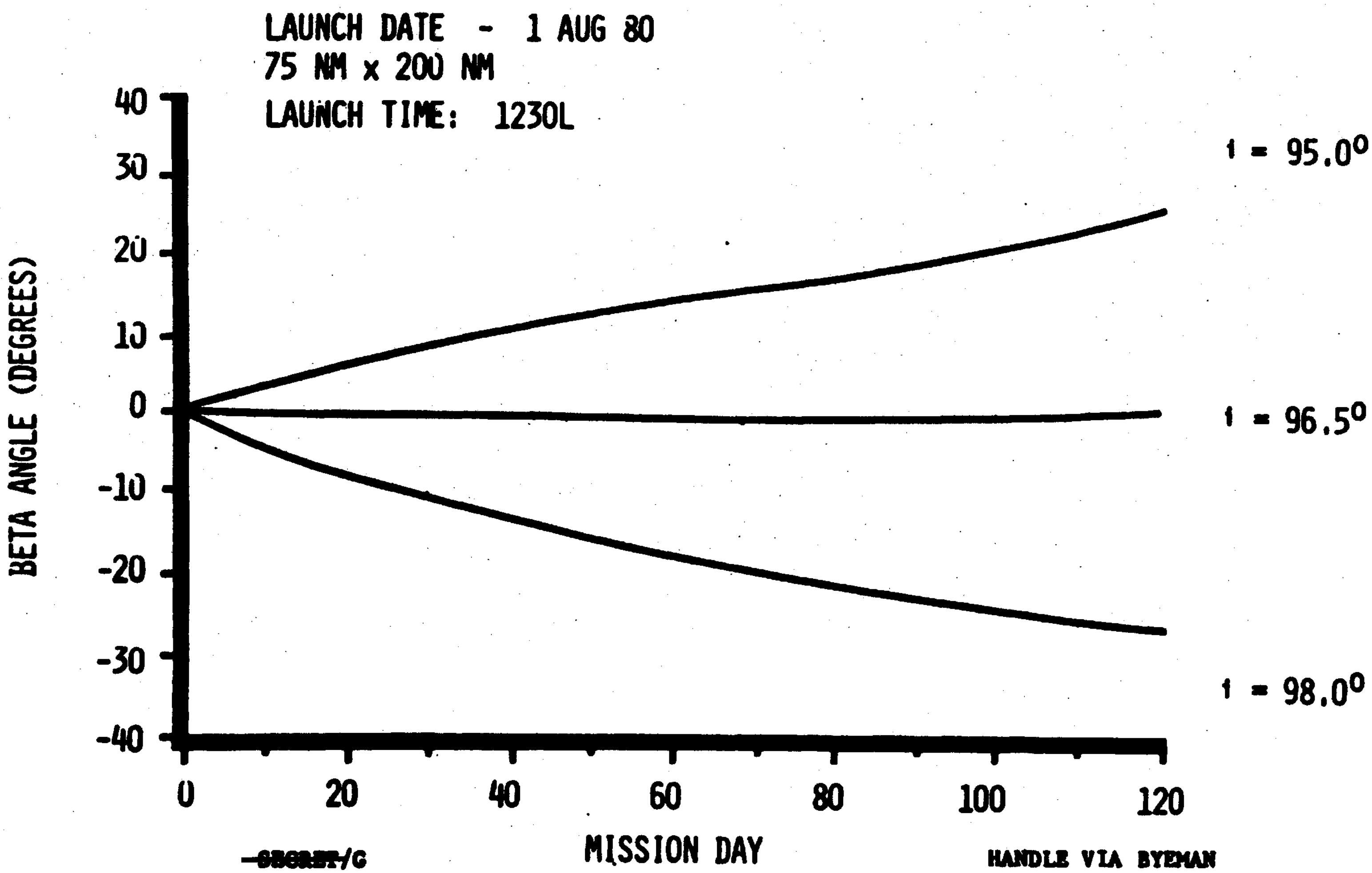

FIGURE 2.1.2 A

PARAMETERS AFFECTING MISSION DURATION

2.1.2 SUN ANGLE (BETA) - Continued

Dual Mode missions will require an injection inclination which is a beta compromise between the fly high and the fly low sun-sync values. This compromise inclination is necessary to maintain a beta within the hardware constraints throughout the mission.

For example, a mission involving a fly low orbit of 75 NM by 200 NM and a fly high orbit of 450 NM circular has a sun-sync inclination associated with each orbit. These are 96.43^{o} and 98.75^{o} respectively. If the mission were flown at either inclination, beta would remain nearly constant for the optimized segment but vary sharply for the non-optimum segment. Since on-orbit inclination changes require extremely large propellant expenditure, it is necessary to choose a compromise inclination. In this case, an inclination of 97.5^{o} provides a near constant value of beta ($\pm 6^{o}$ over 120 days as shown in Figure 2.1.2 B).

The liftoff time or launch window determines the initial value of beta. The conditions which force widely varying beta will require careful selection of launch time if beta is to stay within the thermal and power constraints. This consideration is illustrated in Figure 2.1.2 B for a LO-HI Dual Mode Mission where beta varies at different rates and directions for the two mission segments.

2.1.3 INJECTION INCLINATION - The inclination flown on a specific mission is determined by the vehicle constraints (beta, etc.). The data used in calculating propellant expenditures is based upon a 96.4^{o} inclination. Injection inclinations greater than 96.4^{o} will reduce the on-orbit propellants by approximately 30 lbm per degree offset for small angle differences ($\pm 3^{o}$ differences).

Higher inclination orbits ($\approx 110^{o}$) are possible but at a severe penalty. The longer ground traces over the Area of Interest (AI) provide more photographic opportunities per orbit cycle, however, the propellant requirement to achieve orbit and control argument of perigee, coupled with the rapidly changing sun angles (vehicle and local), prohibit long duration mission.

DUAL MODE ORBIT PLANNING
BETA ANGLE VARIATION

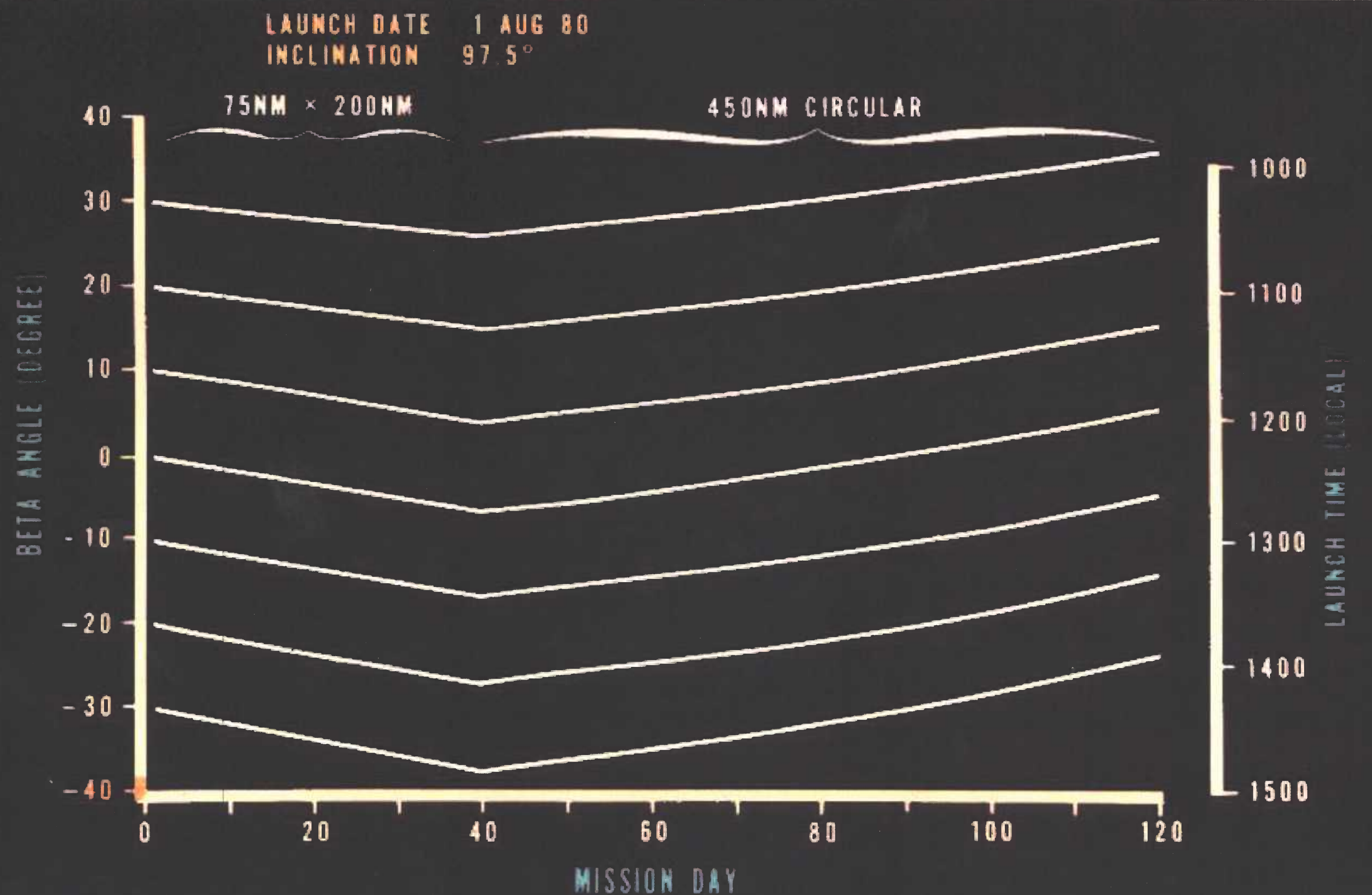

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FIGURE 2.1.2B

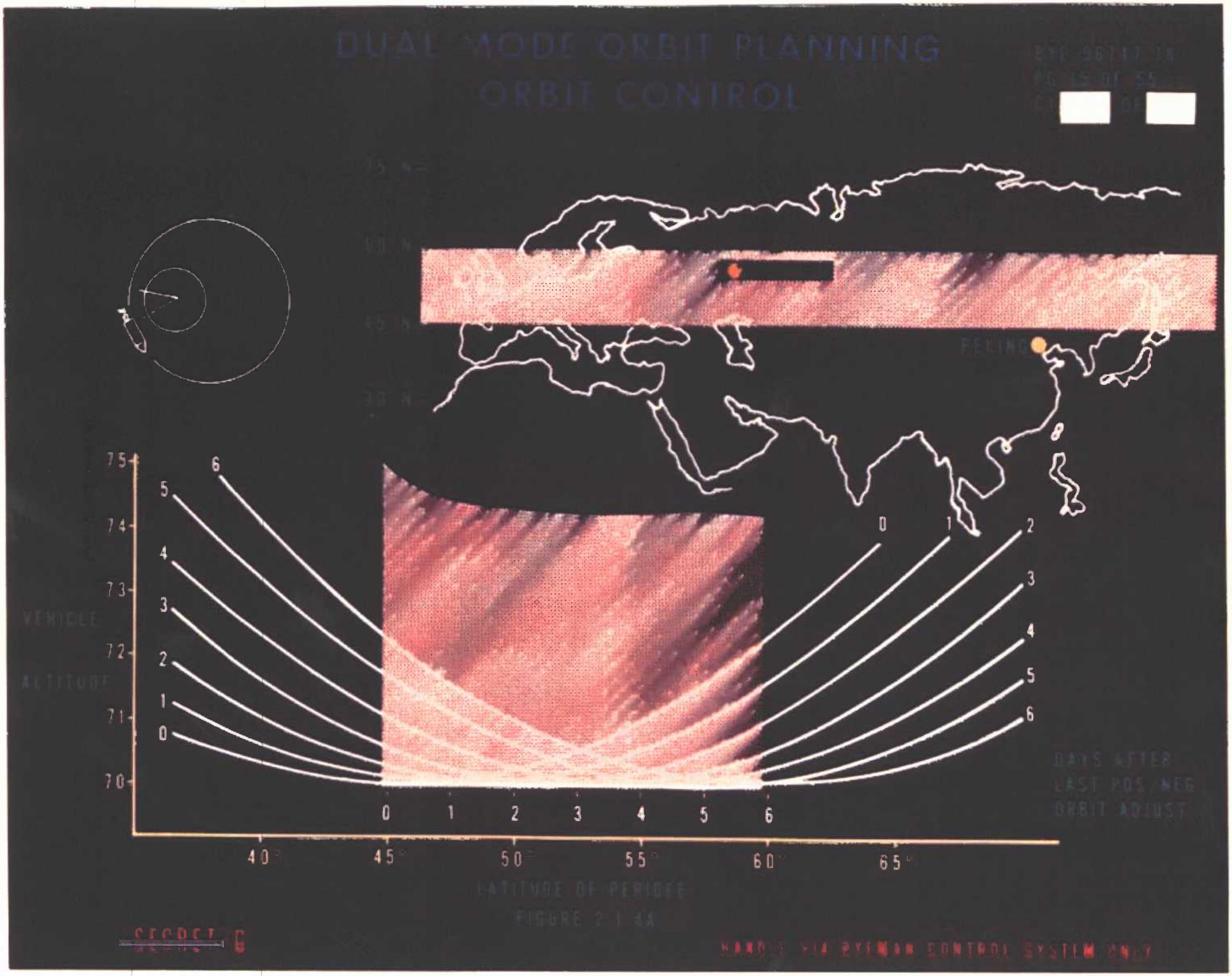
HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

PARAMETERS AFFECTING MISSION DURATION

2.1.4 INJECTION ARGUMENT OF PERIGEE - Historically, the injection trajectory has been designed to place perigee at about 30° N latitude. This was done as a compromise between minimizing the injection propellant requirement and getting the vehicle into the optimum latitude band for high resolution photography over the AI; that is, the 45° N to 60° N latitude band illustrated in Figure 2.1.4. Once on orbit, the first set of OA's makes the final adjustment to 45° N latitude.

Additional injection propellants are used to achieve this orbit because of the extended off-axis burn time required. For short duration missions, this was an acceptable trade-off since it maximized the low altitude duration over the AI. This may be necessary for future fly low missions, however, all Dual Mode missions should, when possible, provide for the selection of an argument of perigee which will maximize the on-orbit propellants. This can be done by designing the injection trajectory to put the vehicle into orbit at or near perigee. This will result in the latitude of perigee being between 10° S and 20° N latitude for sun-sync orbits.

This approach is optimum for Dual Mode orbits which fly at high altitude during the first segment since a transfer orbit adjust is planned shortly after injection. This transfer orbit adjust will put the vehicle into a circular orbit where argument of perigee is not a concern. However, for Dual Mode missions which fly low during the first segment, some consideration must be given to the propellant cost in using the historical approach versus the time required for the latitude of perigee to walk into the desired latitude band.


DUAL MODE ORBIT PLANNING
ORBIT CONTROL
PEKING
VEHICLE
ALTITUDE
75
74
73
72
71
70
0 1 2 3 4 5 6
40° 45° 50° 55° 60° 65°
LATITUDE OF PERIGEE
DAYS AFTER
LAST POS/NEG
ORBIT ADJUST
FIGURE 2.1.8A
~~SECRET~~ G
HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

PARAMETERS AFFECTING MISSION DURATION

2.1.4 INJECTION ARGUMENT OF PERIGEE - Continued

Once on orbit the real world perturbations, such as earth oblateness, will cause the apsidal line (the line between perigee and apogee) to "walk" around the world. The rate at which it walks is a function of inclination, altitude, and the perturbation elements. Figure 2.1.4 B shows the apsidal precession rate as a function of inclination and H-Bar (average altitude) for near circular orbits ($h_a - h_p < 400$ NM).

The following data points illustrate the use of the chart. For a sun-sync orbit of 75 NM by 200 NM ($i = 96.4^o$), the apsidal precession is approximately 4.2 degrees per day northward. A sun-sync 450 NM circular orbit ($i = 98.75^o$) has a walk of approximately 3.5 degrees per day northward. A non-sun-sync 350 NM circular orbit at a 95^o inclination would have an apsidal precession rate of approximately 3.8 degrees per day.

For the case discussed on the previous page, the time to walk into the desired 45^o to 60^o N latitude band from a 10^o N injection point can easily be calculated. Assume a Dual Mode Lo-Hi orbit at a 97.5^o inclination with an injection orbit of 75 NM by 200 NM. The apsidal rate from Figure 2.1.4 B is approximately 4.3 degrees per day. To walk 55 degrees will require 8.5 days (45^o N - 10^o N, 35 degrees/ 4.3 degrees per day = 8.5 days).

ARGUMENT WALK

AS A f(INCLINATION H-BAR)

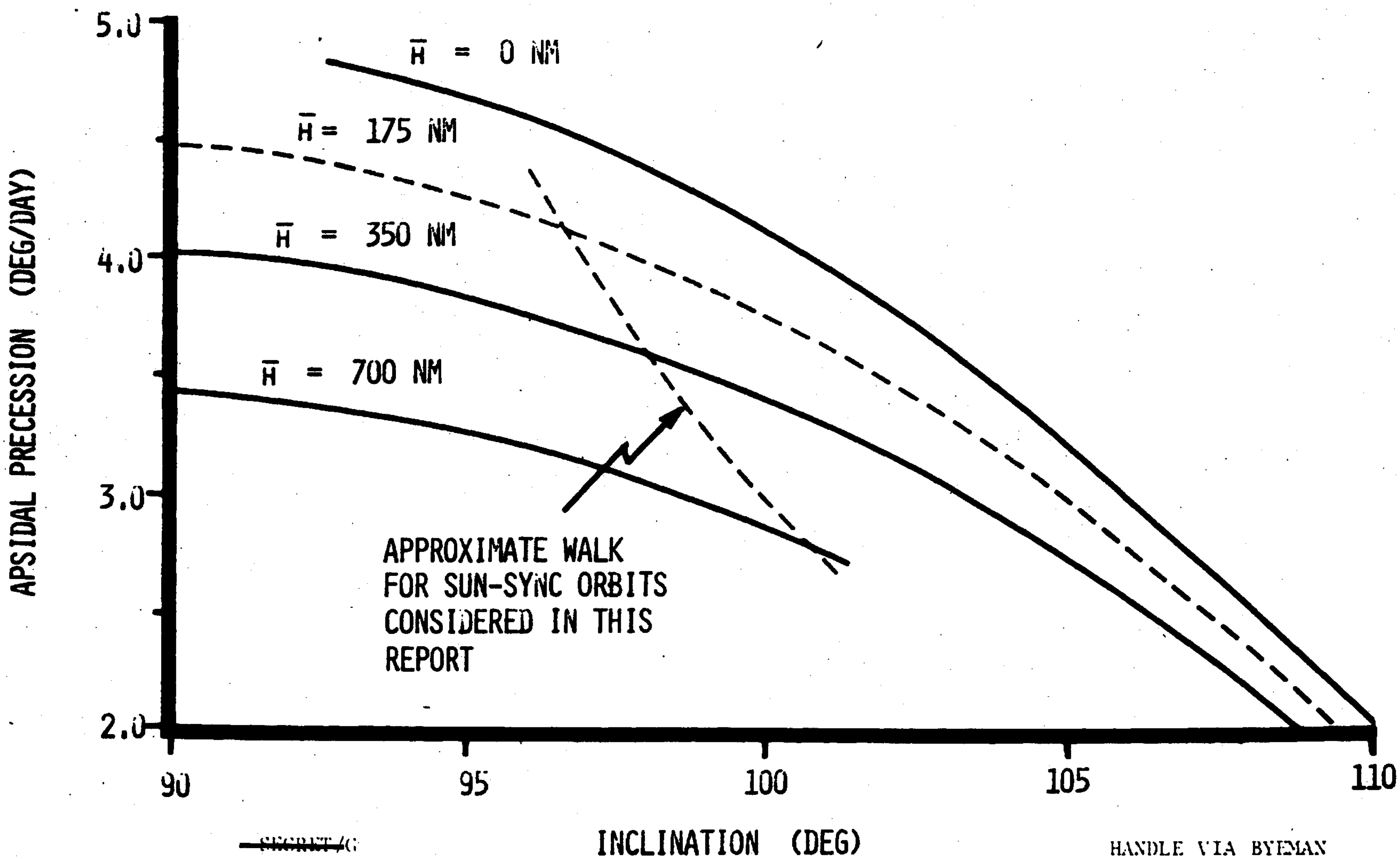


FIGURE 2.1.4B

PARAMETERS AFFECTING MISSION DURATION

2.1.5 INJECTION ALTITUDE PROFILE - The injection orbit altitude profile can vary from the minimum required for adequate orbital life, as defined in the System Test Objective (STO), to a maximum as specified by the Dual Mode mission requirements. The vehicle integrating contractor has performed injection studies for perigees of 75 NM to 85 NM and apogees up to 225 NM. However, apogee altitudes up to 450 NM are not considered limiting since it requires only an additional few seconds of main engine burn (up to ≈ 8 seconds over the current ≈ 220 second burns) to achieve the desired altitude.

The maximum on-orbit propellants can be achieved by injecting into the lowest perigee altitude consistent with orbital lifetime requirements. This will reduce the burn requirements of the sub-orbital injection phase which is accomplished at some alpha angle (α) off the vehicle axis. The missions documented in this report use an injection perigee of 75 NM except where otherwise noted. The propellants remaining at injection versus apogee altitude are shown in Figure 2.1.5 for the nominal 4 battery, 2 hydrazine tank configuration. This data is used throughout this report.

PROPELLANTS REMAINING
AT INJECTION

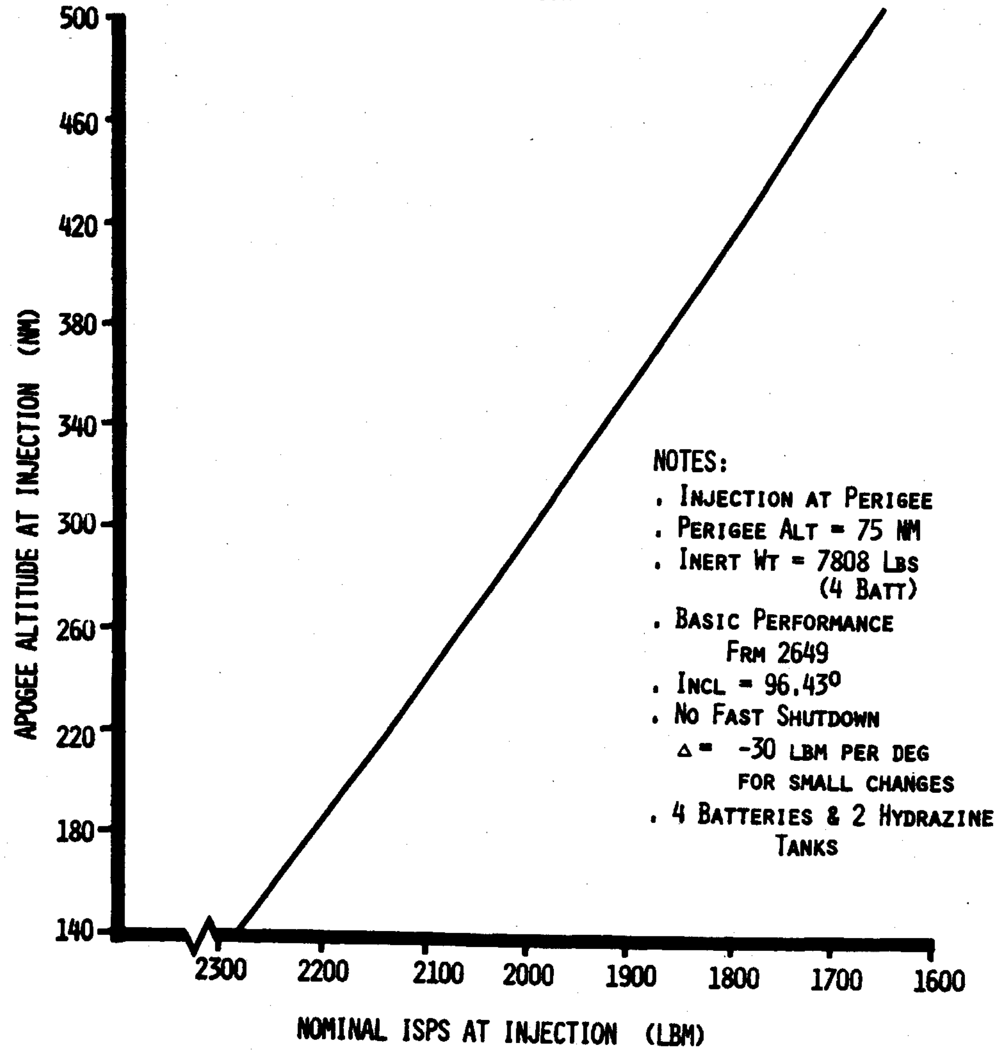


FIGURE 2.1.5

PARAMETERS AFFECTING MISSION DURATION

2.1.6 ARGUMENT OF PERIGEE CONTROL - The largest on-orbit use of propellants is in drage make-up, orbit Q adjustment, and control of the argument (latitude) of perigee. Figure 2.1.4 illustrates a technique used in fly low missions to maintain perigee over the area of interest; that is, 45° N to 60° N latitude. The northward walk of the latitude of perigee can be restricted to about 2.50°/day by the proper placement of daily drag make-up orbit adjusts. However, a positive-negative OA combination is required every six days to bring perigee back to 45° N latitude. Similar techniques are used to provide for positive-negative orbit adjusts on 7, 8, or 9 day cycles.

The amount of propellants required for argument control can be expressed in terms of an average daily velocity delta which must be added to the orbit. This requirement is shown in Figure 2.1.6 for fly low cases having an apogee altitude of approximately 200 NM. Similar curves exist for the more elliptic orbits using perigee argument control in the northern and southern hemispheres.

DUAL MODE ORBIT PLANNING
ORBITAL MAINTENANCE REQUIREMENTS

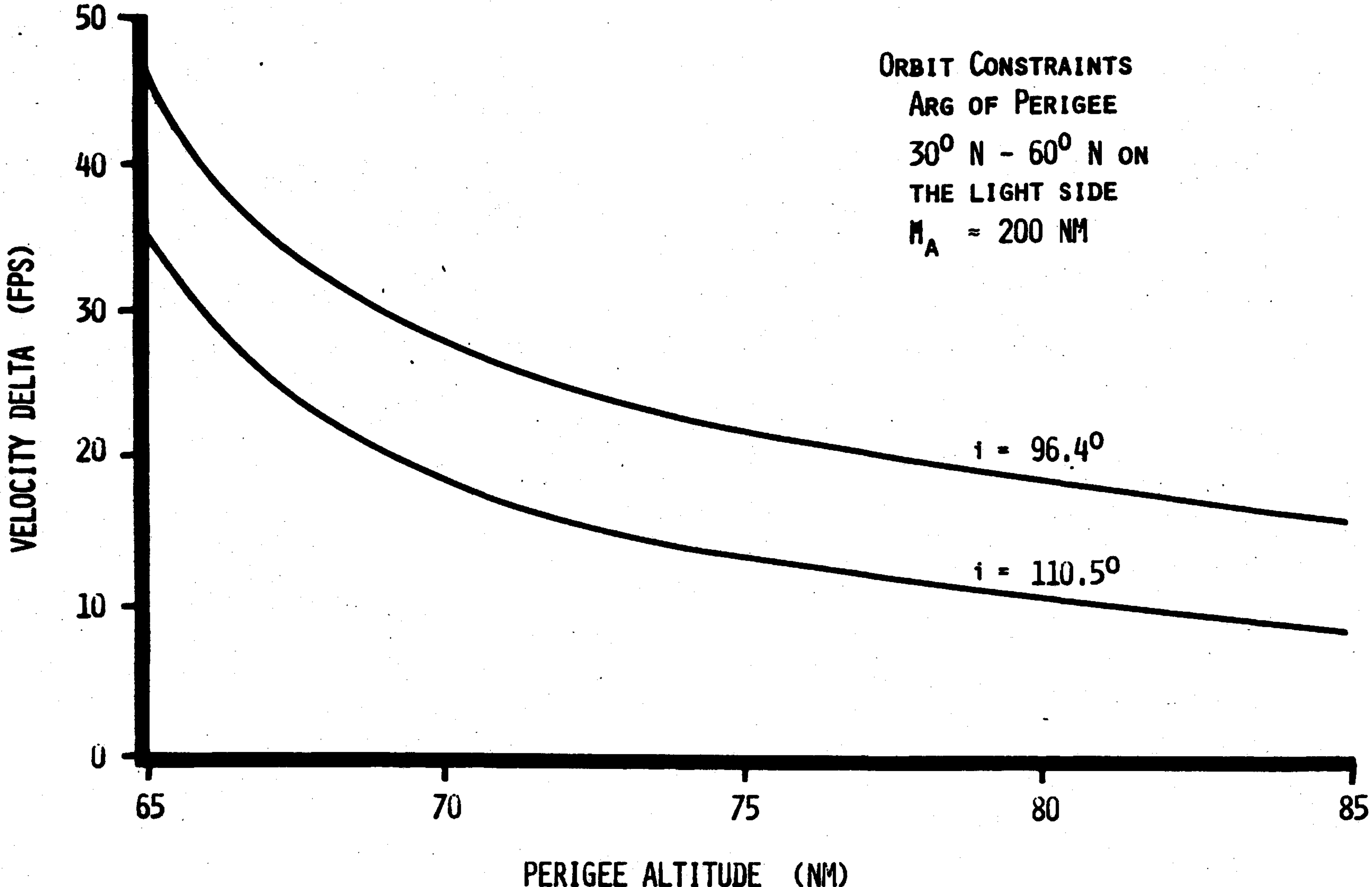


FIGURE 2.1.6

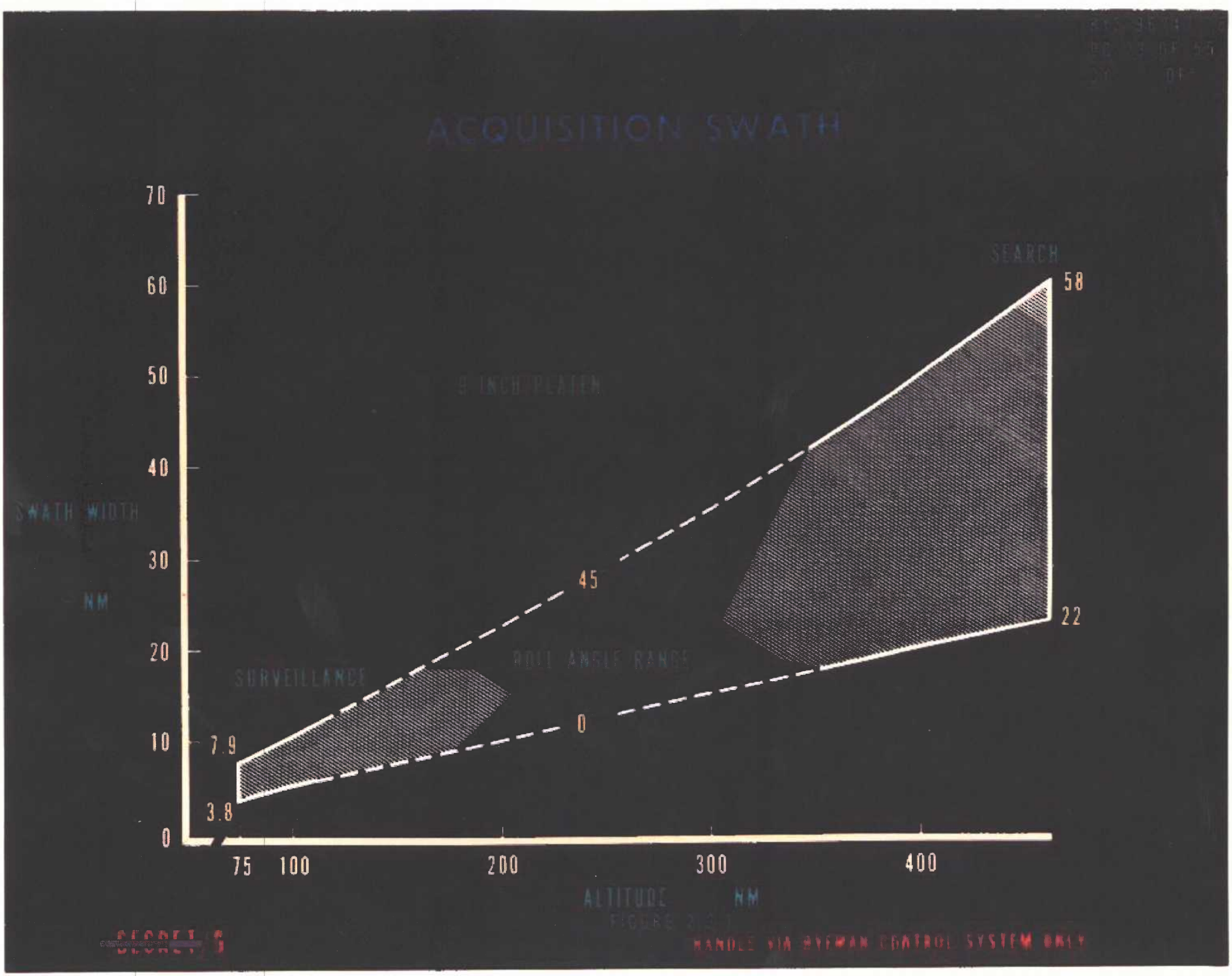
2.2 PARAMETERS AFFECTING ALTITUDE PROFILE - The trade-off of altitude versus mission length is normally decided by mission requirements and ISPS propellants available on orbit. However, several other factors must be considered.

2.2.1 AREA ACCESS - The fly high altitude profiles require that the film drive speed electronics be modified to provide a second speed range which is approximately one-fourth the current capability. This new speed range will provide photographic capability at slant ranges from 65 to 710 nautical miles.

The maximum target area access can be achieved only by utilizing the full capability of the roll joint. This is best illustrated in Figure 2.2.1. Retaining current obliquity options (roll, crab, stereo) will restrict the maximum fly high altitude to approximately 450 nautical miles. Higher altitudes are possible but only by limiting the roll joint capability. Therefore, the 450 nautical mile altitude is considered a hardware limitation and is observed throughout this report.

2.2.2 RECOVERY DISPERSIONS - The in-track dispersions expected for the Dual Mode vehicle configuration are dependent upon four major factors. These are orbit perigee, argument of perigee, SRV retrorocket thrust, and impulse variation. Acceptable dispersions can be achieved with the present TEM-236 retro-rocket motor for circular orbits up to approximately 400 NM and for elliptic orbits of any argument of perigee with apogee below 400 NM. This is based on sun-synchronous orbits, 1.1 percent maximum impulse variation, in-track dispersion limit of 270 NM and nominal CG offset on the SRV. If apogee is allowed to increase to 450 NM, then the argument of perigee must be constrained to a range of 90^{o} to 290^{o} to limit the in-track dispersions.

If the proposed new, higher thrust SRV rocket motor is purchased and if the impulse variation is less than 1.5 percent, then circular and elliptic orbits up to 450 NM apogee, without argument of perigee constraints, can be achieved within the 270 NM in-track dispersion limit.


ACQUISITION SWATH
SWATH WIDTH
NM
70
60
50
40
30
20
10
0
SEARCH
58
22
8 INCH PLATEN
45
0
ROLL ANGLE RANGE
SURVEILLANCE
7.9
3.8
75
100
200
300
400
ALTITUDE NM
SECRET
HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

PARAMETERS AFFECTING ALTITUDE PROFILE

2.2.3 HDA CONSTRAINT - The ISPS propellant tank (Figure 1.0A) is a single unit with a diaphragm which separates the hypergolic oxidizer and fuel. Any rupture of this diaphragm would cause instantaneous combustion of the propellants and destruction of the vehicle.

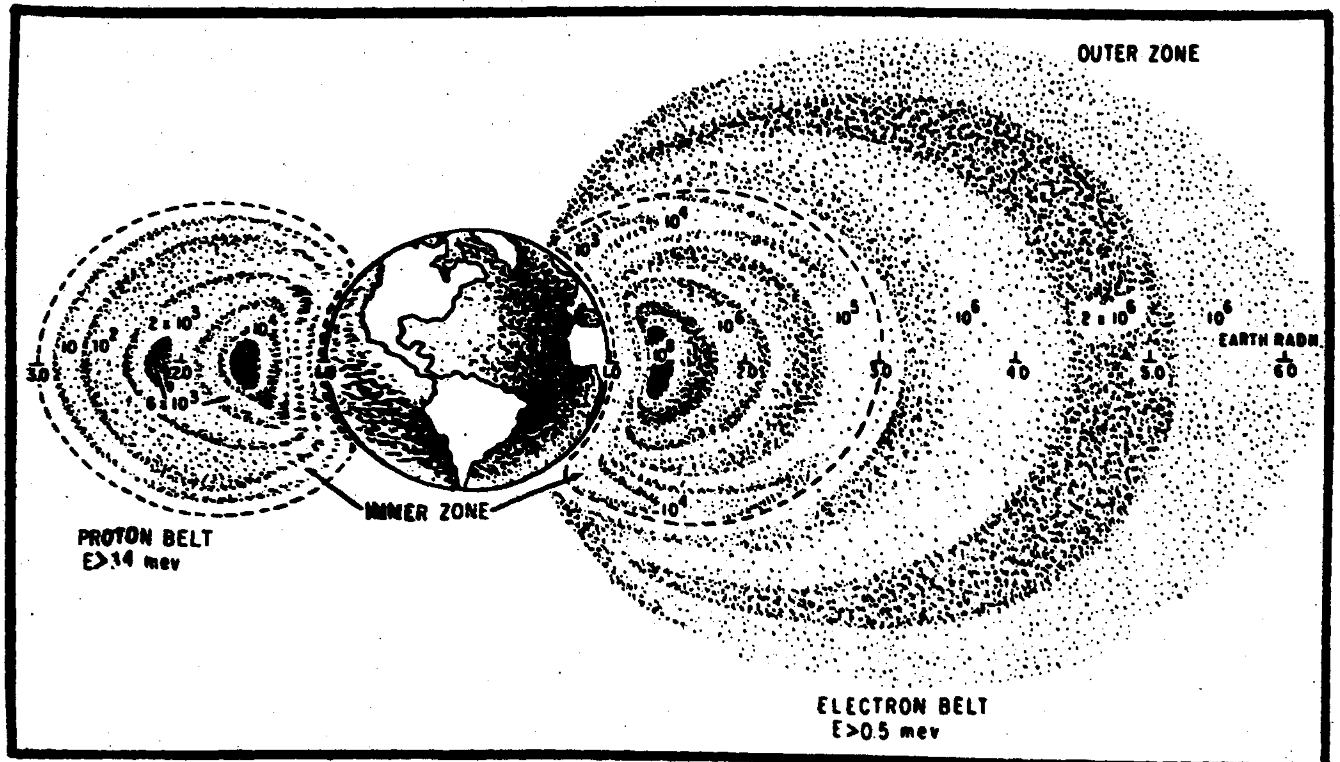
The high saturated vapor pressure of the nitrogen tetroxide constituent of the HDA oxidizer is a critical factor limiting the aerodynamic heat that the oxidizer tank can absorb. The aerodynamic heat absorbed by the oxidizer tank is small if the vehicle is aligned with the orbital velocity vector, straight and level as in normal flight, because the angle of attack and the tank area exposed to the relative wind are very small. Hence, the only time that aerodynamic heating becomes a concern is when the altitude of the vehicle results in a high angle of attack and large tank area exposure. This can occur if the attitude control of the vehicle is lost or when executing a planned attitude maneuver is performed.

The concern over possible increased oxidizer tank pressure leading to a diaphragm reversal and rupture has led to the current HDA constraint. This constraint is based upon simplified assumptions; the vehicle is flying broadside (sideways) through perigee, a 1962 standard atmosphere, and the composition of the oxidizer liquid and vapor remain invariant as the mass of oxidizer vapor increases (tank heats up). The full development of the constraint is beyond the scope of this text but the threat is real and the constraint is observed through a set of curves. These curves, three, four, and five rev constraint lines, define the maximum number of tumbling revs the vehicle can survive before detonation. These are a function of altitude flown and the real atmospheric environment.

Generally speaking, flying the vehicle above 85 NM poses very little threat to the vehicle due to the low density of the atmosphere. Also, the first half of the mission is relatively safe since a larger mass of oxidizer liquid is available to absorb any heat input. However, late in the mission when the mass of oxidizer is low, particularly when the vehicle is at or below 75 NM, the constraint plays a major role in how the vehicle is flown. A loss of attitude control during periods of fluctuating atmospheric conditions can create catastrophic results. Therefore, adequate precautions are taken to insure the vehicle can be recaptured and stabilized within the defined constraint period. This often becomes a difficult problem due to multiple dead revs (no station contact), lack of redundant systems due to prior failure, or unexpected severe solar activity causing drastic increases in the atmospheric drag.

PARAMETERS AFFECTING ALTITUDE PROFILE

2.2.4 SPACE RADIATION ENVIRONMENT - Naturally occurring and artificially injected charged particles are trapped in the earth's magnetic fields in what are normally called the "Van Allen Belts." These belts are the product of a number of significant events such as the high altitude nuclear tests of 1952, solar storm particle discharges, high energy bombardment of the atmosphere by cosmic radiation, and the polar auroral dumping of very low energy electrons. These events produce electrons, protons, and helium nuclei of various energy levels which are distributed in space as shown below.



2.2.4 SPACE RADIATION ENVIRONMENT - Continued

One of these belts consists primarily of energetic protons which are stably trapped in the earth's inner zone. These protons cycle between the poles in a spiral motion about constant lines of magnetic flux. A low flying satellite (apogee below 500 NM) will encounter these protons in the region of the South Atlantic Magnetic Anomaly. This anomaly can be viewed as a "sag" in the magnetic flux field intensity and its accompanying energetic protons. This is pictorially shown in Figure 2.2.4.

A photographic satellite vehicle passing through this anomaly may experience a "fogging" of its film due to an accumulation of this radiation (energetic protons). Successive passes through this region will increase the total dosage until the film becomes partially exposed or "fogged." The amount of radiation induced on a given cycle is a function of solar or cosmic activity, the shielding provided by the vehicle, the susceptibility of the film to radiation, and the path taken through the region. The solar or cosmic activity is an independent, uncontrollable variable. However, the path taken through the region is controllable provided the remaining variables are well understood.

Tests are in progress to determine the vehicle shielding effects and how best to control the film exposure to this radiation. Once this is determined, altitude limitations or orbital planning restrictions may be imposed.

DUAL MODE ORBIT PLANNING
"THE RADIATION PROBLEM"

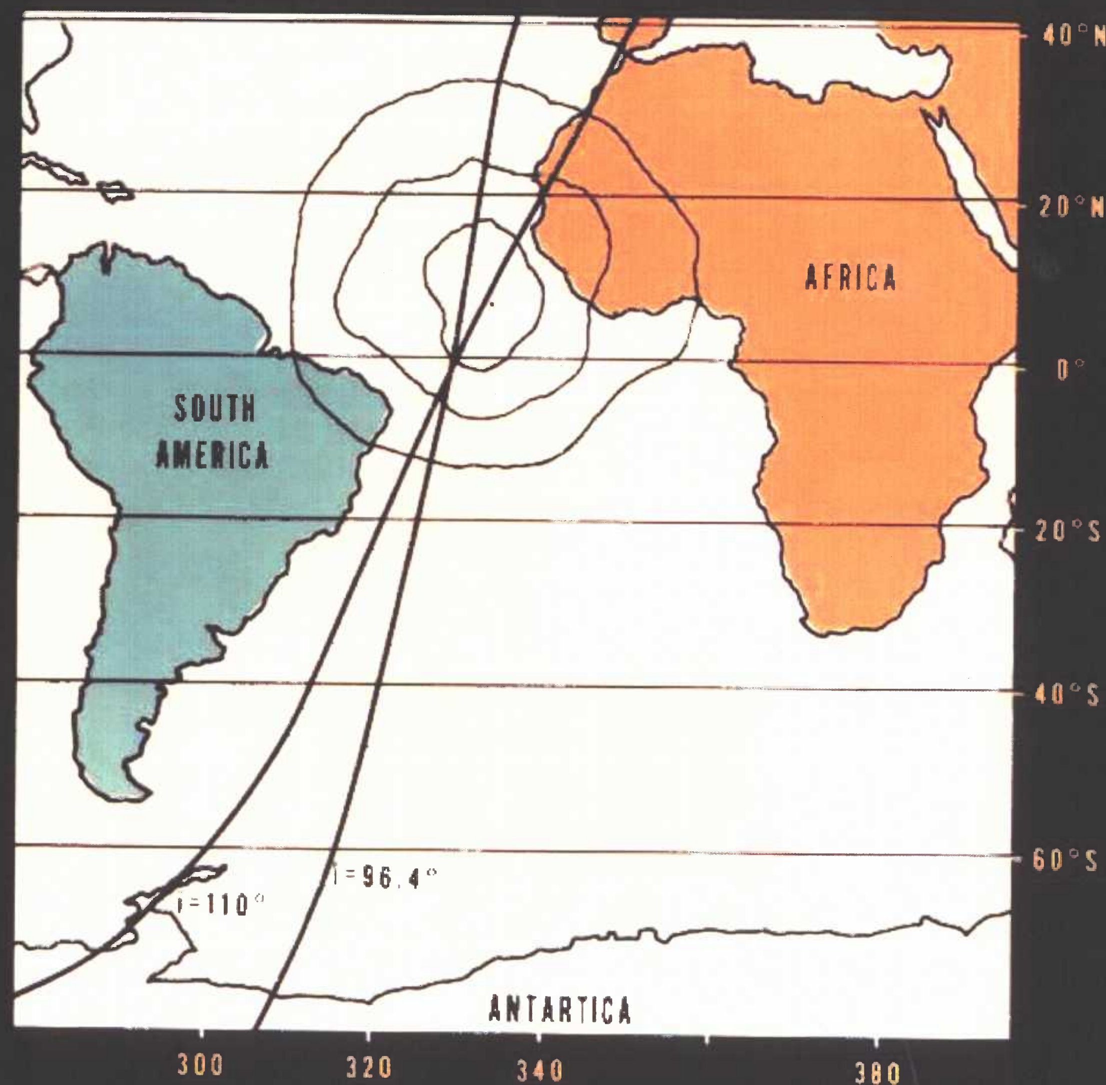


FIGURE 2.2.4

INFLUENCING FACTORS

- SOLAR ACTIVITY
- PERIGEE HEIGHT
- APOGEE HEIGHT
- ARGUEMENT OF PERIGEE
- INCLINATION
- ORBIT Q

~~SECRET~~/G
HANDLE VIA BYEMAN
CONTROL SYSTEM ONLY

PARAMETERS AFFECTING ALTITUDE PROFILE

2.2.5 ATMOSPHERIC DRAG - The drag or deceleration force on a satellite vehicle as it moves through the upper atmosphere is described in part by the aerodynamic relationship

$$\frac{D}{M} = \frac{1}{2} \left[\frac{C_D}{M} \quad A \right] \rho \left[V^2 \right]$$

where the drag coefficient C_D, cross-sectional area A, and mass M are vehicle characteristics. The density term ρ describes the atmospheric characteristics as a function of altitude. The vehicle velocity V is a function of the orbit.

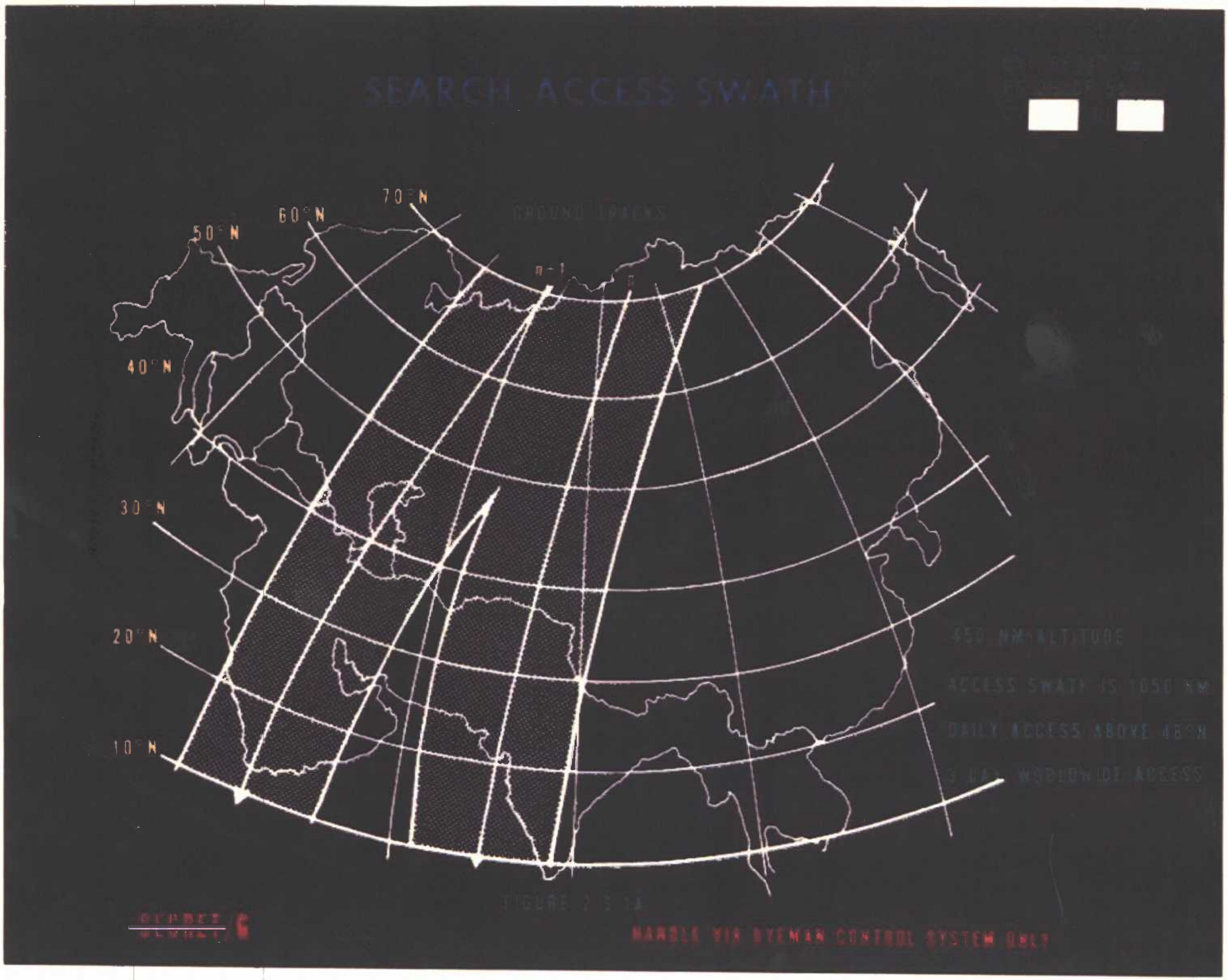
At altitudes above 85 NM the density term, and thus the atmospheric drag, becomes negligible. However, for lower altitude regimes, particularly below 75 NM, high atmospheric drag removes orbital energy (vehicle local velocity) and allows the vehicle altitude to decay. Maintaining a defined orbit; i.e., 75 NM x 200 NM, requires that the orbital energy be maintained constant at the value associated with that orbit. This requires periodic ISPS orbit adjust burns to replace the energy lost to drag which increases propellant expenditures and forces a trade-off between mission length and altitude profile.

Solar activity (flares, etc.) creates unexpected increases in the molecular activity of the atmosphere. This activity increases the drag on the vehicle and requires higher than expected ISPS propellant expenditure to maintain the orbit characteristics. Obviously propellant utilization becomes one of the principal on-orbit controlled resources.

2.3 ADDITIONAL FACTORS - Several additional factors directly influence the final selection of the parameters discussed in Sections 2.1 and 2.2.

2.3.1 ORBIT Q SELECTION - Several strategies can be developed for fly high and fly low orbit profiles. For fly low cases presented in this report, the previously established strategies have been used since the data is readily available.

The fly high orbit profiles result in ground swaths over the AI similar to those shown in Figure 2.3.1 A. The selection of orbit Q will determine how many days are required to achieve a 100 percent world wide access; that is, reduce the critical latitude to zero.


SEARCH ACCESS SWATH
GROUND TRACKS
70°N
60°N
50°N
40°N
30°N
20°N
10°N
n+1
n
450 NM ALTITUDE
ACCESS SWATH IS 1050 KM
DAILY ACCESS ABOVE 48°N
3 DAY WORLDWIDE ACCESS
SECRET/G
HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

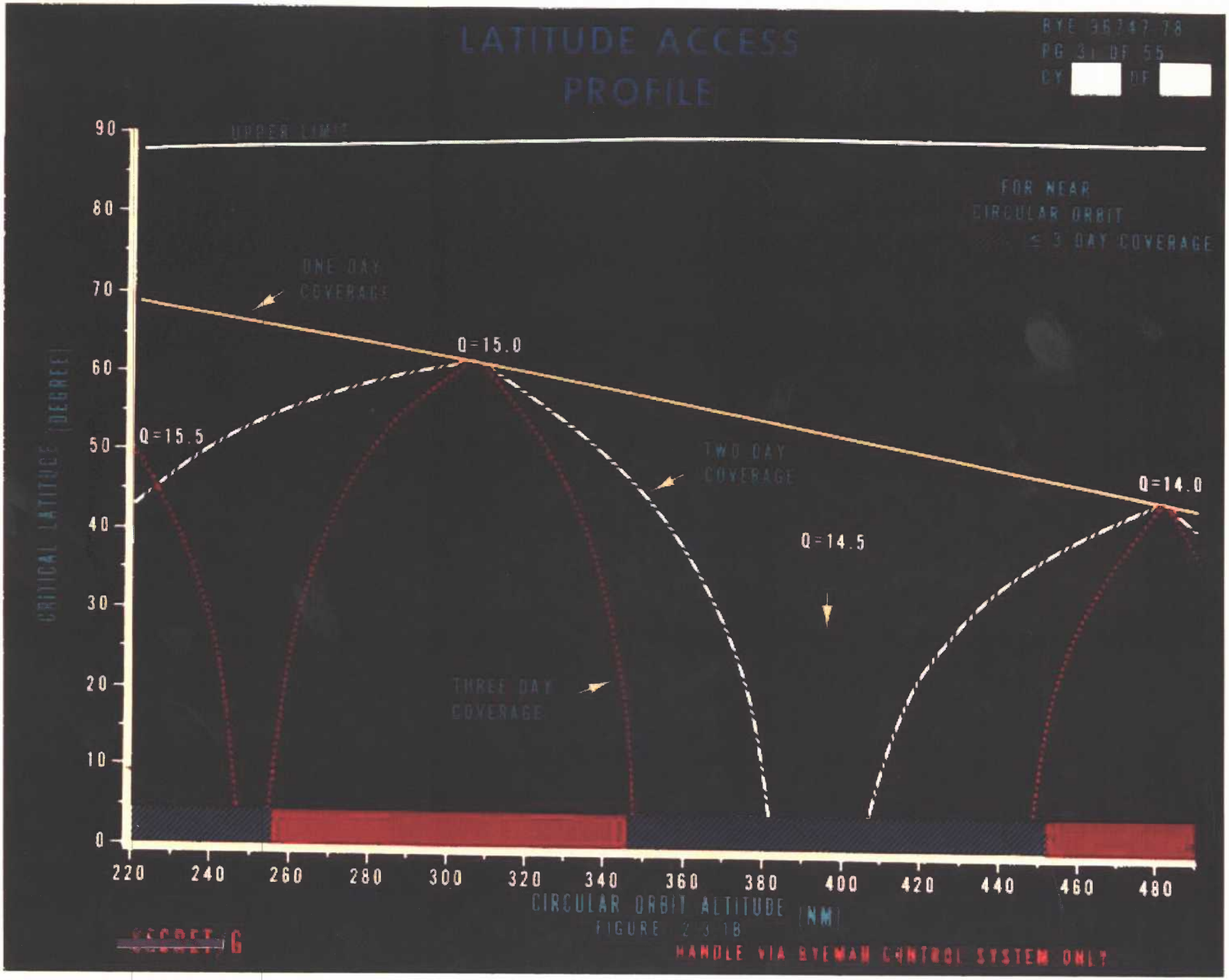
ADDITIONAL FACTORS

2.3.1 Cont'd - The critical latitude is defined as the lowest latitude at which overlapping ground swaths provide 100 percent world-wide coverage. For the example shown in Figure 2.3.1A, the critical latitude is 48° N for one day access. The selection of orbit Q will determine the rate at which the ground traces walk around the world. The increased overlap on successive days can reduce the critical latitude to zero thus providing 100 percent world-wide access. The relationship of critical latitude to orbit Q for near circular orbits is shown in Figure 2.3.1B.

The following example illustrate the use of this chart. A circular orbit of 360 NM can provide 100% access from a northern limit of approximately 89° N down to a lower limit of: 1 day - 57° N, 2 day - 39° N, 3 day - 0° N. This orbit would provide 100% world wide coverage in three (3) days.

For this report, access coverage of up to three days has been arbitrarily designated as desirable. All others can be supported, and may even be desirable for specific conditions, but are shown here as undesirable. This criteria is based on an assumption of maximizing clear weather opportunities over the area of interest. From Figure 2.3.1B it is possible to determine the range of $\bar{h}$ (average or circular altitude) of orbits which will provide 100 percent access of the earth in three days or less.

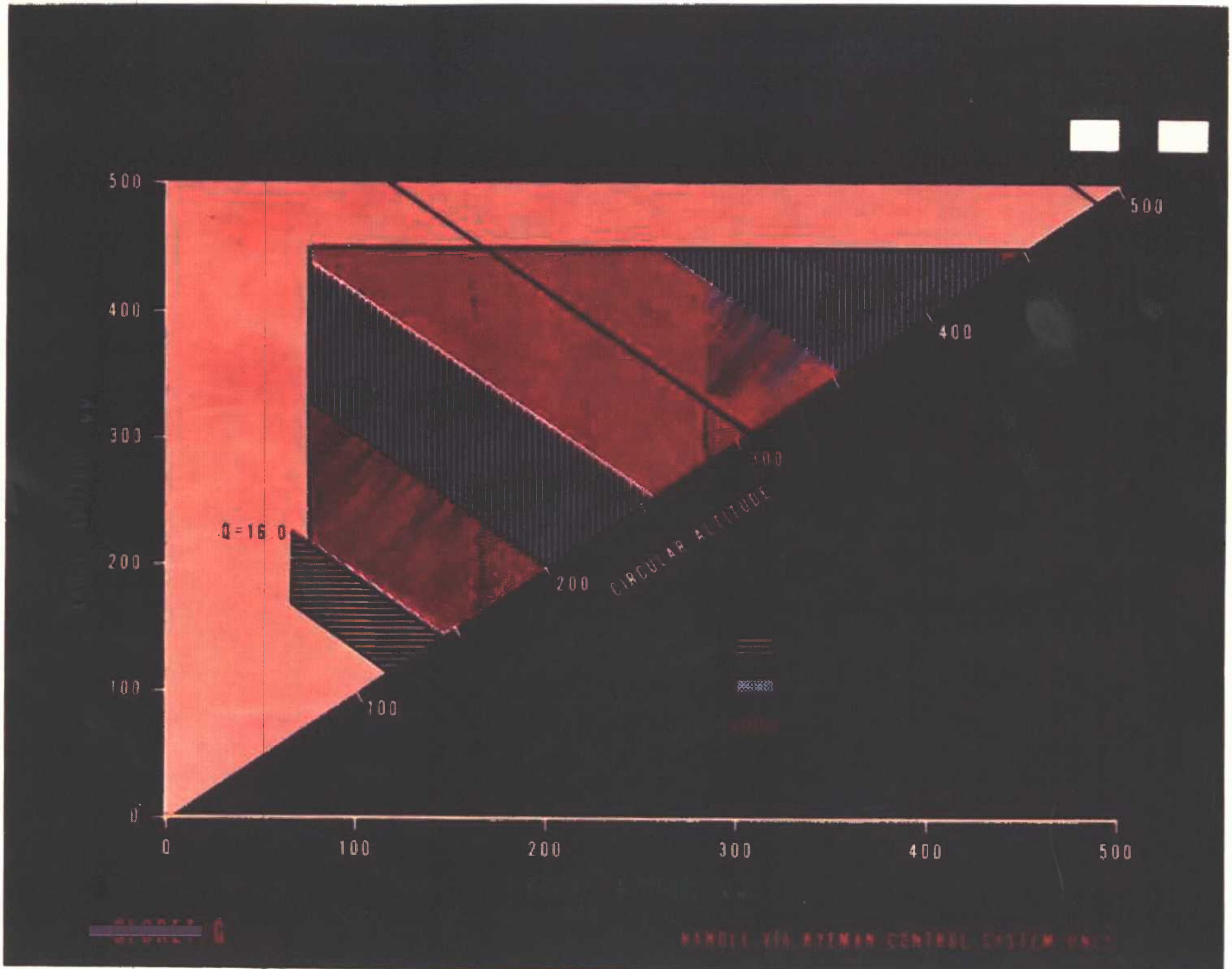
100% access ≤ 3 days	100% access > 3 days
220 - 255 NM	255 - 345 NM
345 - 450 NM	450 - 490 NM


LATITUDE ACCESS PROFILE
BYE 36747 78
PG 31 OF 55
CY OF
UPPER LIMIT
FOR NEAR CIRCULAR ORBIT ≤ 3 DAY COVERAGE
ONE DAY COVERAGE
TWO DAY COVERAGE
THREE DAY COVERAGE
Q=15.0
Q=15.5
Q=14.5
Q=14.0
CRITICAL LATITUDE (DEGREE)
90
80
70
60
50
40
30
20
10
0
220
240
260
280
300
320
340
360
380
400
420
440
460
480
CIRCULAR ORBIT ALTITUDE (NM)
FIGURE 2.3.1B
~~SECRET~~/G
HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

ADDITIONAL FACTORS

2.3.1 Cont'd - The orbit Q and coverage cycles for near circular elliptic orbits ($h_a - h_p < 400$ NM) having apogee over the area of interest can be determined from Figure 2.3.1C (approximation). For example, a 200 NM by 300 NM orbit with apogee maintained over the area of interest has a Q of approximately 15.6 and a 100 percent coverage cycle of < 3 days--a desirable access cycle.

NOTE: The fly low altitude profiles shown in Figure 2.3.1C have perigee over the area of interest.


500
400
300
200
100
0
Q=16.0
CIRCULAR ALTITUDE
100
200
300
400
500

ADDITIONAL FACTORS

2.3.2 ISPS CORROSION CONSTRAINT - The ISPS engines are normally used for all orbit adjusts. The corrosive nature of the HDA oxidizer reacts with the engine plumbing to creat a corrosion by-product which can clog the thruster injectors or filters and render the engine unusable (Ref. Figure 2.3.2). An inhibitor is added to the HDA to retard this corrosion, however, adequate protection cannot be completely obtained without serious impact to the specific impulse. Therefore, a constraint exists which requires that the engines be burned a minimum of ten seconds every six days (if propellants are in the engine plumbing) to purge partially formed corrosion by-products. This constraint has been observed in the data presented in this report.

2.3.3 USE OF THE MAIN ENGINE (ME) - For Hi-Lo Dual Mode missions, it is possible to leave the ISPS engines dry and use the ME for the transfer orbit. This would eliminate the need for ISPS burns at fly high circular altitude to observe the ISPS corrosion constraint.

Studies show that the ME offers propellant savings when used for orbit adjust burns requiring greater than 600 lbm of propellants. This performance improvement is due to the higher specific impulse of the ME (300 vs 267 for the ISPS). However, there is an associated increase in the orbit dispersion data resulting from the tail off in thrust during shutdown and from the higher 3 sigma thrust variation. These dispersions are acceptable when the transfer orbit is not performed in conjunction with the injection orbit.

For orbit adjust burns using less than 600 lbm propellants, the high tail off losses at shutdown negate any specific impulse performance gains.

AGENA MPS/ISPS CONFIGURATION

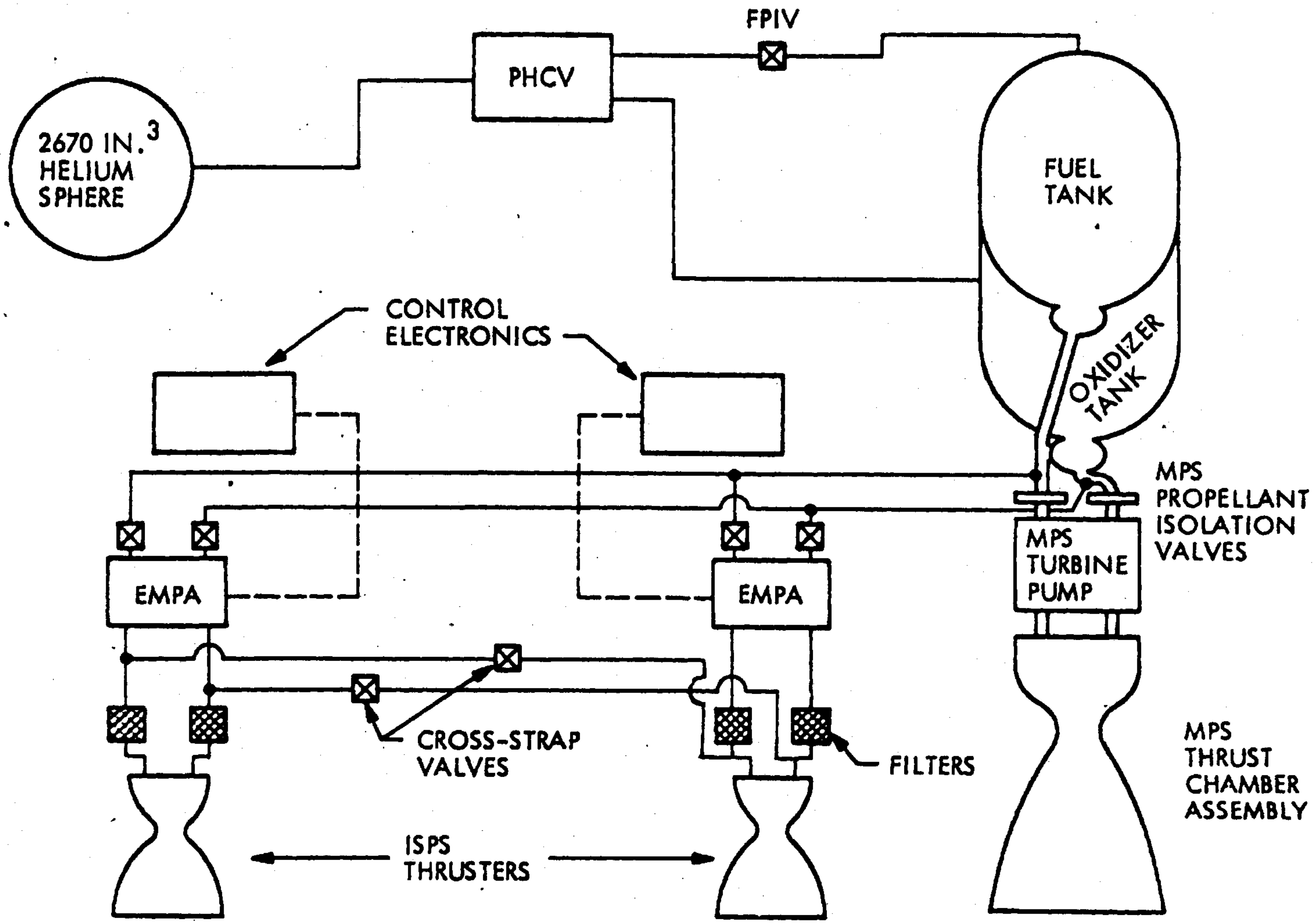


FIGURE 2.3.2

DUAL MODE FLY LOW

3.0 FLY LOW ONLY MISSIONS - The vehicle modifications required to provide a GAMBIT Dual Mode capability will not impact the current operational capability. This capability is shown in Figure 3.0 and was developed using the following criteria and assumptions:

1) Inject into a 75 NM x 200 NM orbit

2) Recover SRV-1 at mid-mission

3) Recover SRV-2 at end of mission

4) ISPS engines used on all orbit adjusts and deboost

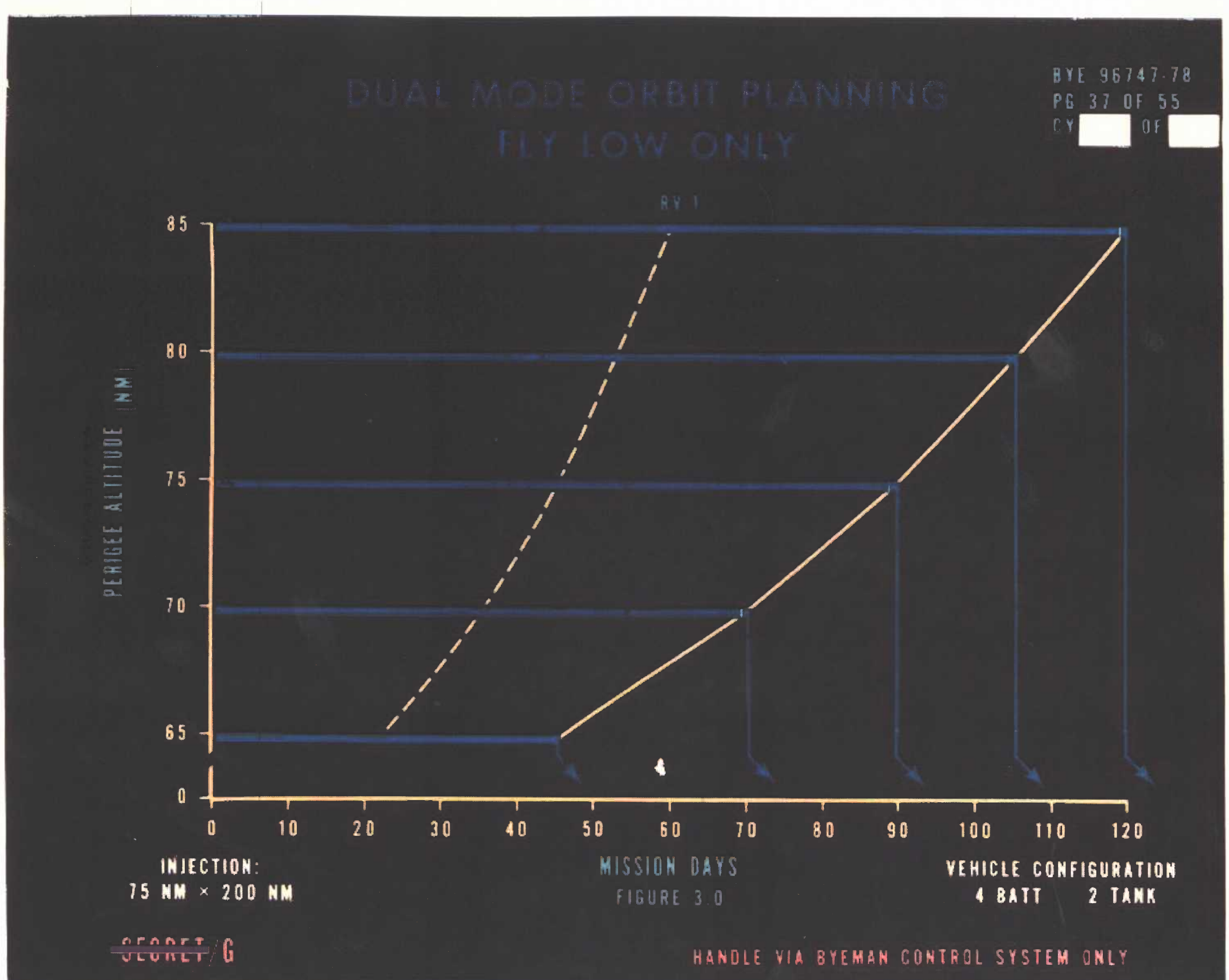
5) Maintain a 150 lbm propellant reserve to allow for vapor and uncertainty (not operationally usable).

6) Four batteries and two hydrazine tanks

7) Sun-sync inclination ($\approx 96.5^{\circ}$)

8) Argument control - 45° to 60° N latitude band

The capability of flying 70 mission days at an average perigee altitude of 70 NM was demonstrated on Vehicle 49. The HDA constraint will require that altitude profiles below 70 NM provide a gradual step up in perigee altitude toward the end of mission (Section 2.2.3).


BYE 96747-78
PG 37 OF 55
CY OF
DUAL MODE ORBIT PLANNING
FLY LOW ONLY
RV 1
85
80
75
70
65
0
PERIGEE ALTITUDE (NM)
0 10 20 30 40 50 60 70 80 90 100 110 120
INJECTION:
75 NM × 200 NM
MISSION DAYS
FIGURE 3.0
VEHICLE CONFIGURATION
4 BATT 2 TANK
~~SECRET~~/G
HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

DUAL MODE FLY HIGH

4.0 FLY HIGH ONLY MISSIONS - The injection phase will be designed to maximize on-orbit propellants. The injection orbit perigee will be at 75 NM and located at or near the injection latitude (10° S - 10° N). Apogee will be the desired final fly high altitude; i.e., 75 NM x 450 NM. The final orbit is achieved by a series of orbit adjust burns using the ME or ISPS engines. The sequence of these OA burns can vary widely depending upon the mission scenario chosen. Generally speaking, none of these sequences will feed back any impact on the injection trajectory which would cause it to be different from the normal type of ascent.

The fly high only capability is summarized in Figure 4.0. It was developed based upon the same criteria and assumptions listed in Section 3.0. However, the sun-sync inclination varies with circular altitude flown (i.e., 450 NM equates to 98.75° sun-sync inclination). In addition:

1) Mission length - 120 days

2) Circular fly high orbit ($h_a - h_p \approx 10$ NM)

3) Corrosion constraint observed at high altitude - 10 sec. burn every six days ($\approx$ 50 lbm propellants)

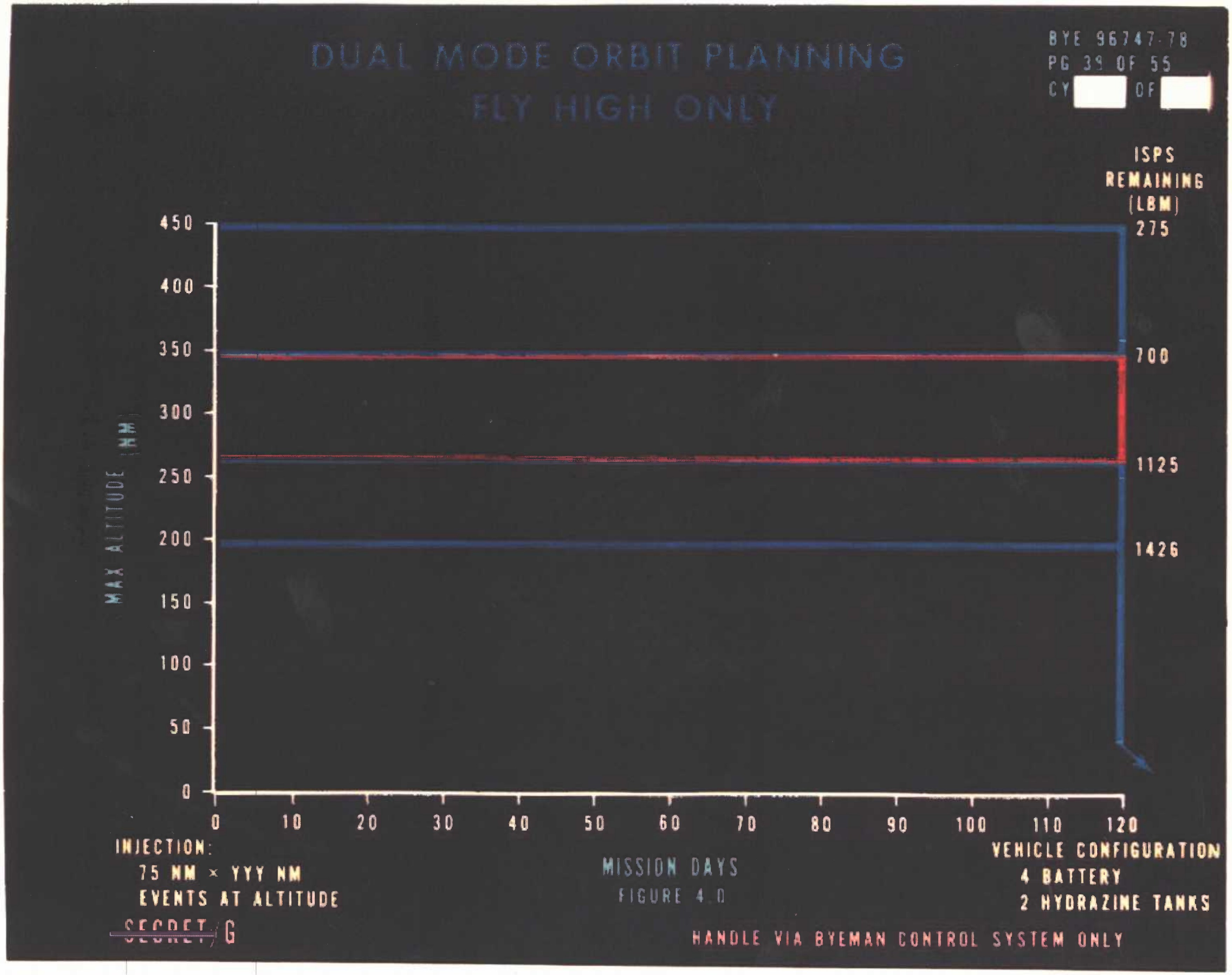
4) SRV's recovered from fly high altitude

The value of ISPS remaining as shown in Figure 4.0 is the amount of propellants in the tanks upon deboost impact. These propellants are available for use in:

1) ASAT contingencies

2) Extending the mission beyond 120 days, provided electrical power is available. Any additional battery requirements would reduce this reserve. The additional ISPS required for the corrosion constraint is small, less than 5 percent of total on-orbit ISPS available for extensions of up to 60 days.

3) Flying low at the end of mission prior to SRV-2 recovery



DUAL MODE ORBIT PLANNING
FLY HIGH ONLY
BYE 96747-78
PG 39 OF 55
CY OF
ISPS REMAINING (LBM)
275
700
1125
1426
MAX ALTITUDE (NM)
450
400
350
300
250
200
150
100
50
0
0 10 20 30 40 50 60 70 80 90 100 110 120
MISSION DAYS
FIGURE 4.0
INJECTION:
75 NM × YYY NM
EVENTS AT ALTITUDE
VEHICLE CONFIGURATION
4 BATTERY
2 HYDRAZINE TANKS
~~SECRET~~/G
HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

HI-LO DUAL MODE MISSIONS

5.0 HI-LO DUAL MODE MISSIONS - The injection orbit and fly high planning for the HI-LO Dual Mode orbit is identical to the fly high only case (Section 4.0). However, a fly low segment is provided by using the excess propellants shown in Figure 4.0. This requires expenditure of propellants to lower the altitude and to control the argument of perigee within the desired 45^{o} N to 60^{o} N latitude band. Two fly low profiles are possible: 1) lower perigee only and perform a "dive bomb" fly low mission, or 2) lower perigee and apogee and conduct a standard fly low segment. The development of both profiles is based on the same criteria and assumptions found in Section 3.0 with the following additions:

1) Fly high segment terminated at 80 days
2) SRV-1 recovered at fly high altitude (< day 80)
3) Circular fly high orbit ($h_a - h_p \approx 10$ NM)
4) Injection inclination of 97.5^{o} (See Section 2.1.2)
5) Corrosion constraint observed at fly high altitude - 10 sec. burn every six days (≈ 35 lbm propellants)

5.1 FLY LOW PERIGEE ONLY - This assumes a negative OA at the planned apogee to lower perigee, and daily OA's to control argument of perigee within the 45 - 60^{o} N latitude band. The average daily propellant expenditure is calculated using the average daily delta velocity data developed in Section 2.1.6. Figure 5.1 defines the HI-LO Dual Mode capability for this option. The following examples illustrate how to use the chart:

1) Inject into a 75 NM by 400 NM orbit, circularize to 400 NM, and then step down after 80 days to a 75 NM by 400 NM fly low orbit. The maximum fly low duration, from Figure 5.1, is 14 days. The fly high access closure would be less than three days as derived from Figure 2.3.1 A, a desirable factor as defined in Section 2.3.1. The fly high closure and orbit Q can be approximated from Figure 2.3.1 B (2 days and 14.4 respectively).

2) Inject into a 75 NM by 300 NM orbit, circularize to 300 NM, and then step down after 80 days to an 80 NM by 300 NM fly low orbit. The maximum fly low duration is 45 days with a fly high closure of greater than 3 days. This closure is not desirable as defined in Section 2.3.1. The fly high orbit Q is 15.1.

These orbits would have to be evaluated for acceptability based upon current targeting strategies. The trade-offs involved include selecting a fly high circular altitude which, without changing apogee, can be transformed into a fly low orbit having an acceptable orbit Q. This evaluation will not be addressed in this report.

DUAL MODE ORBIT PLANNING
HI - LO CAPABILITY

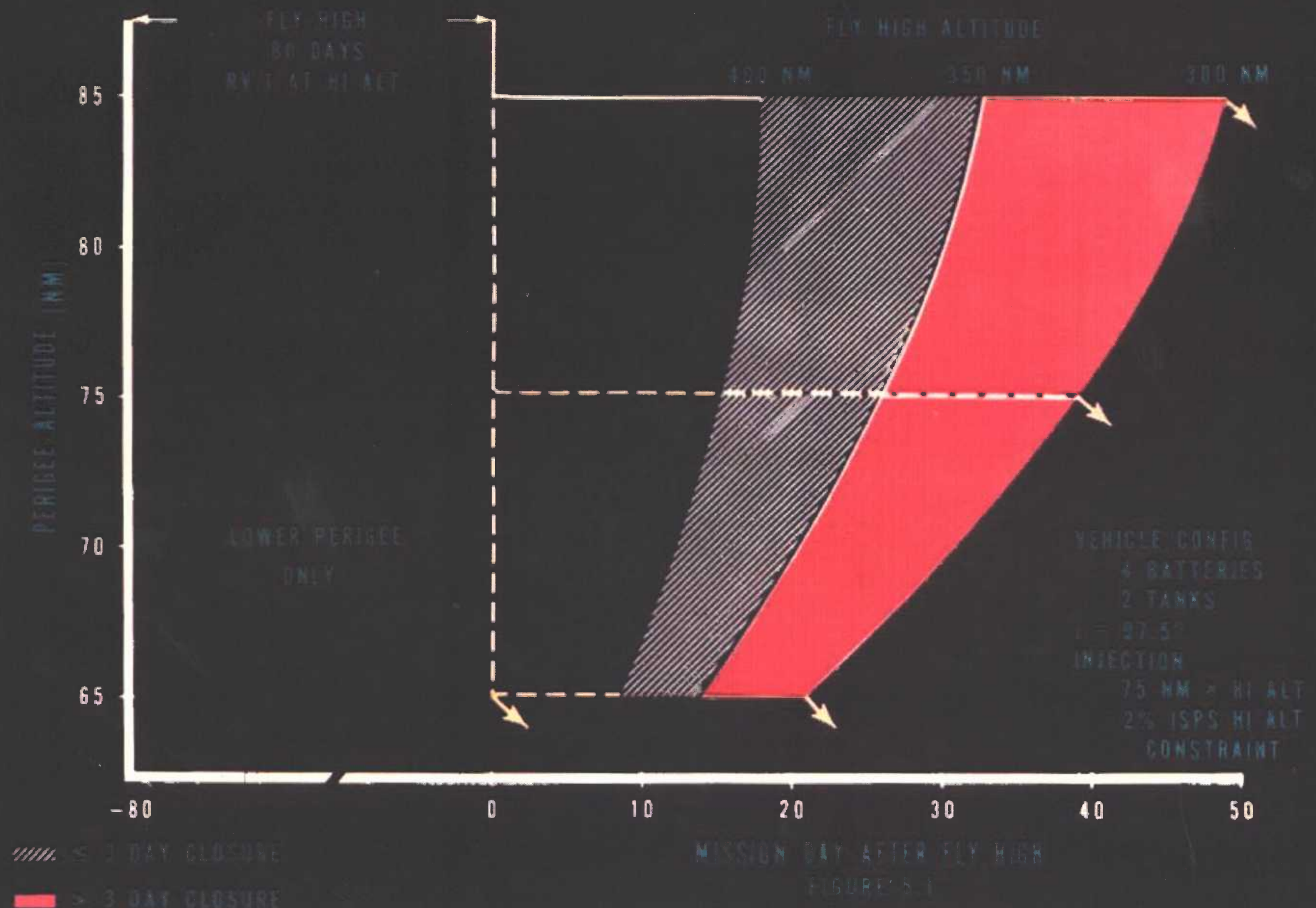


FIGURE 5.1

SECRET G

HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

HI-LO DUAL MODE MISSIONS

5.2 FLY LOW PERIGEE AND APOGEE - This requires a second negative OA at perigee of the fly low orbit described in Section 5.1 to bring apogee altitude down to 200 NM. The argument of perigee control requires slightly less ISPS propellant. Figure 5.2 defines the HI-LO Dual Mode capability for this option. The chart is used in the same manner as in Section 5.1. For the examples used earlier, the following fly low observations can be made:

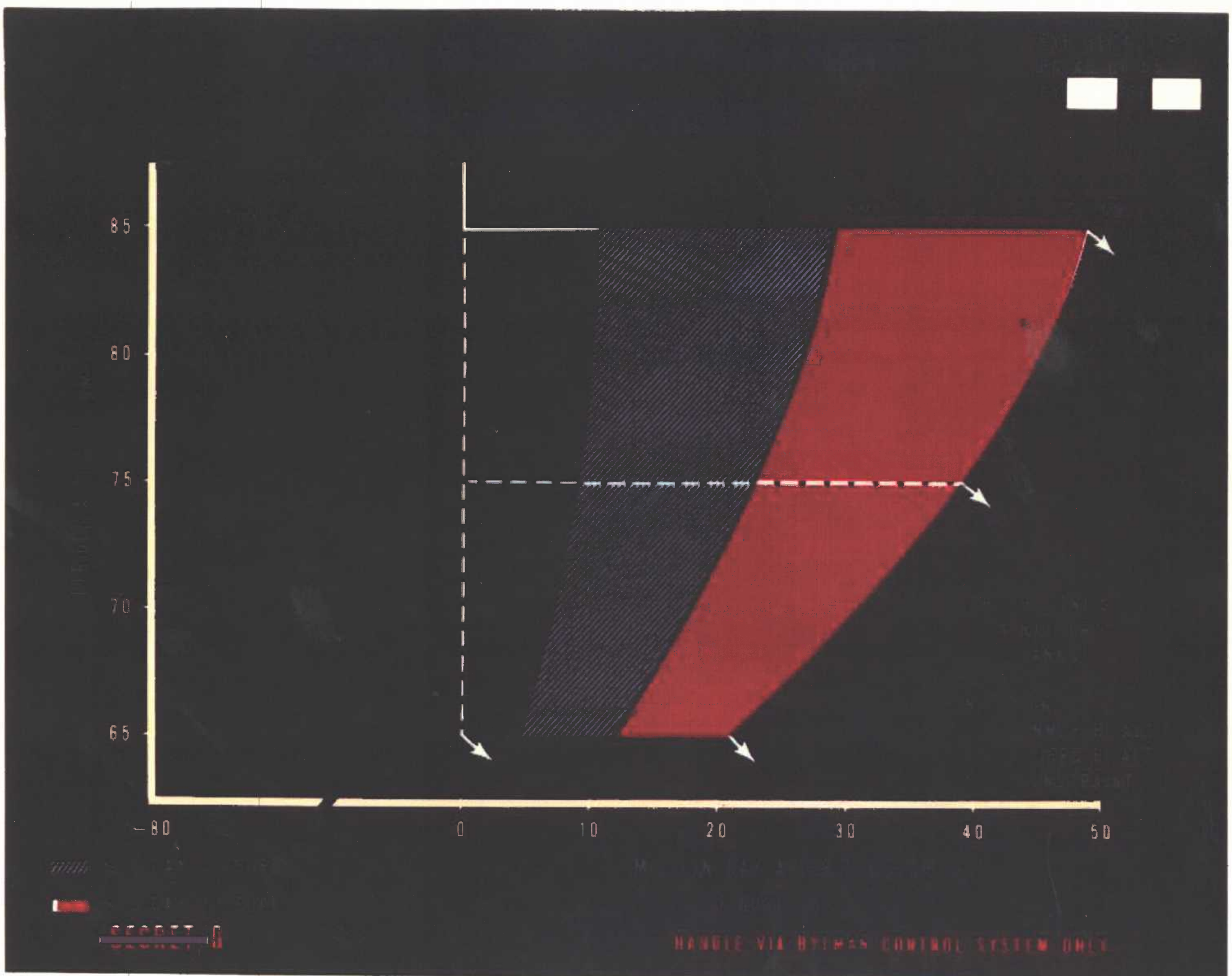
1) The 400 NM circular orbit followed by a 75 NM by 200 NM orbit will allow 9 days of fly low. The fly high Q remains desirable (less than 3 days).

2) The 300 NM circular orbit followed by an 80 NM by 200 NM orbit will allow 45 days of fly low. The fly high Q remains undesirable (greater than 3 days).

The fly high segment is not restricted to 80 mission days in either option. Extension is definitely possible. For example, an additional 60 mission days at high altitude would require an additional ISPS engine constraint propellant expenditure of less than 30 lbm reserved from the propellants available at injection. However, additional power requirements may require more batteries, thus reducing ISPS available (Section 2.1).

The fly low apogee (200 NM) is not a constraint, only a mean value for the range of orbit Q's flown in the past. The specific apogee value will be a function of the inclination, orbit Q, and perigee altitude defined for the specific mission. The delta in ISPS required to fly at a different apogee altitude is negligible for the small altitude difference involved.

The capability to fly extended low altitude profiles at the end of mission is limited by the additional weight carried into the fly high orbit. This weight is comprised of the SRV (and associated film) and the ISPS propellants needed to provide the fly low segment. Performing the fly low segment first will increase the capability of the Dual Mode vehicle.


85
80
75
70
65
−80
0
10
20
30
40
50
SECRET
HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

LO-HI DUAL MODE MISSIONS

6.0 LO-HI DUAL MODE MISSIONS - Substantially longer fly low mission life can be achieved by performing the fly low segment first. The propellant savings is obtained by transferring into a higher orbit with a lighter vehicle (propellant weight loss and possible SRV-1 recovery) and from the fact that it is only necessary to decrease perigee for deboost from the fly high altitude at end of mission.

6.1 LO-HI PROFILE - The injection phase will be identical to the fly low only case except that the inclination will be a compromise between the two sun-sync conditions. Once on orbit, the desired fly low portion is performed using propellants at a rate derived from Section 2.1.6. SRV-1 can be recovered prior to or after transfer into the fly high orbit. After the fly high segment is complete, SRV-2 is recovered from this altitude and the vehicle deboosted. The LO-HI Dual Mode capability is defined in Figure 6.1 and is derived based upon the same criteria and assumptions found in Section 3.0 with the following additions:

1) Fly high segment of 80 days

2) Recover SRV-1 prior to transfer to high orbit

3) Circular fly high orbit ($h_a - h_p \approx 10$ NM)

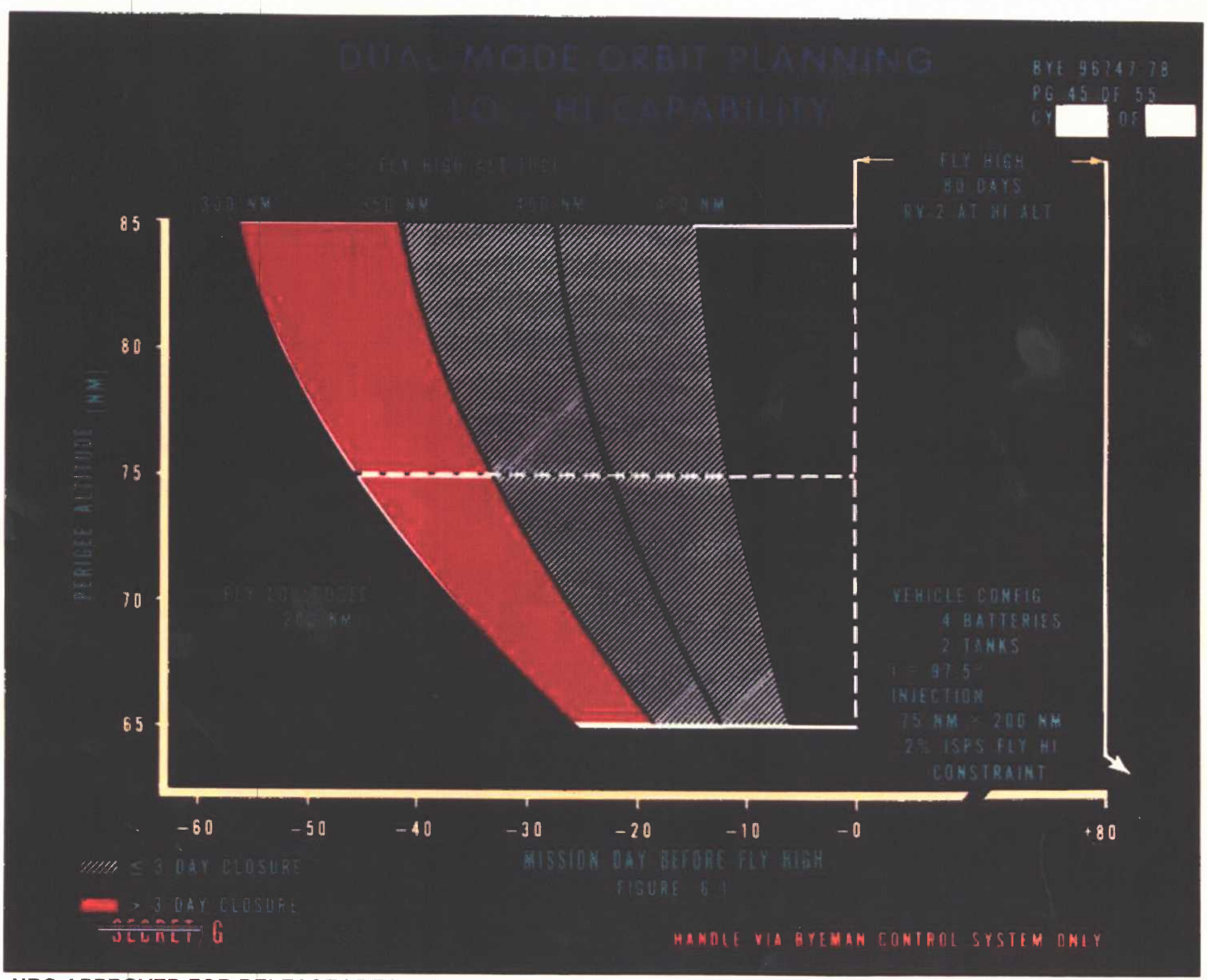
4) Injection inclination of 97.5^{o} (See Section 2.1.2)

5) Corrosion constraint observed at fly high altitude -10 sec. burn every six days ($\approx$ 35 lbm propellants)

The following example illustrates how to use Figure 6.1:

Inject into a 75 NM by 200 NM orbit, fly low at 70 NM by 200 NM for AAA days, recover SRV-1, then circularize to 350 NM circular, fly 80 days, recover SRV-2, and deboost. From Figure 6.1, the fly low capability, AAA, is found to be approximately 27 days. Referring to Section 2.3.1, the fly high Q is 14.75 with a desirable access coverage of $\leq$ 3 days.

Recovery of SRV-1 prior to transfer reduces the weight carried into the fly high orbit. The propellant savings can be equated to approximately 1 or 2 days of added fly low life or to $\approx$ 10 miles increase in the fly high altitude. This trade-off would be based on mission requirements which are not addressed here.


DUAL MODE ORBIT PLANNING
LO - HI CAPABILITY
BYE 96747-78
PG 45 OF 55
CY OF
FLY HIGH
80 DAYS
RV-2 AT HI ALT
300 NM
350 NM
400 NM
85
80
75
70
65
PERIGEE ALTITUDE (NM)
VEHICLE CONFIG
4 BATTERIES
2 TANKS
i = 97.5°
INJECTION
75 NM × 200 NM
2% ISPS FLY HI
CONSTRAINT
-60
-50
-40
-30
-20
-10
-0
+80
MISSION DAY BEFORE FLY HIGH
FIGURE 6-1
≤ 3 DAY CLOSURE
> 3 DAY CLOSURE
SECRET G
HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

LO-HI DUAL MODE MISSIONS

6.2 LO-HI-LO OPTION - A short fly low "dive bomb" surveillance segment can be accomplished at the end of the fly high segment as shown in Figure 6.2. This requires a negative OA to lower perigee as described in Section 5.1 and to place the argument of perigee at 30° N latititude (or lower if mission planners desire). The argument of perigee would be allowed to walk to 70° N at its natural apsidal rate. This rate, determined from Figure 2.1.4 B to be approximately 4.0 degrees per day, would permit a six to ten day fly low segment (10 days if perigee starts at 30° N, six if perigee starts at 45° N). The additional propellants required to carry the SRV into the "dive bomb" orbit (vs deboost) can be made up by flying the earlier segments slightly different (i.e., fly low at a slightly higher altitude). The acceptability of the orbit Q during this "dive bomb" segment would have to be evaluated based upon mission requirements. Any trade-offs to improve this Q may compromise the acceptability of the previous LO-HI segments. Extension beyond 120 days will require a reassessment of power requirements.

The fly high segment is not restricted to 80 mission days in either option. Extension is definitely possible. For example, an additional 60 mission days at high altitude would require an additional ISPS engine constraint propellant expenditure of less than 30 lbm reserved from the propellants available at injection. However, additional power requirements may require more batteries, thus reducing ISPS available on orbit (Section 2.1).

The fly low apogee (200 NM) is not a constraint, only a mean value for the range of orbit Q's flown in the past. The specific apogee value will be a function of the inclination, orbit Q, and perigee altitude defined for the specific mission. The delta in ISPS required to fly at a different apogee altitude is negligible for the small altitude difference involved.

DUAL MODE ORBIT PLANNING
FLY LO-HI-LO OPTION

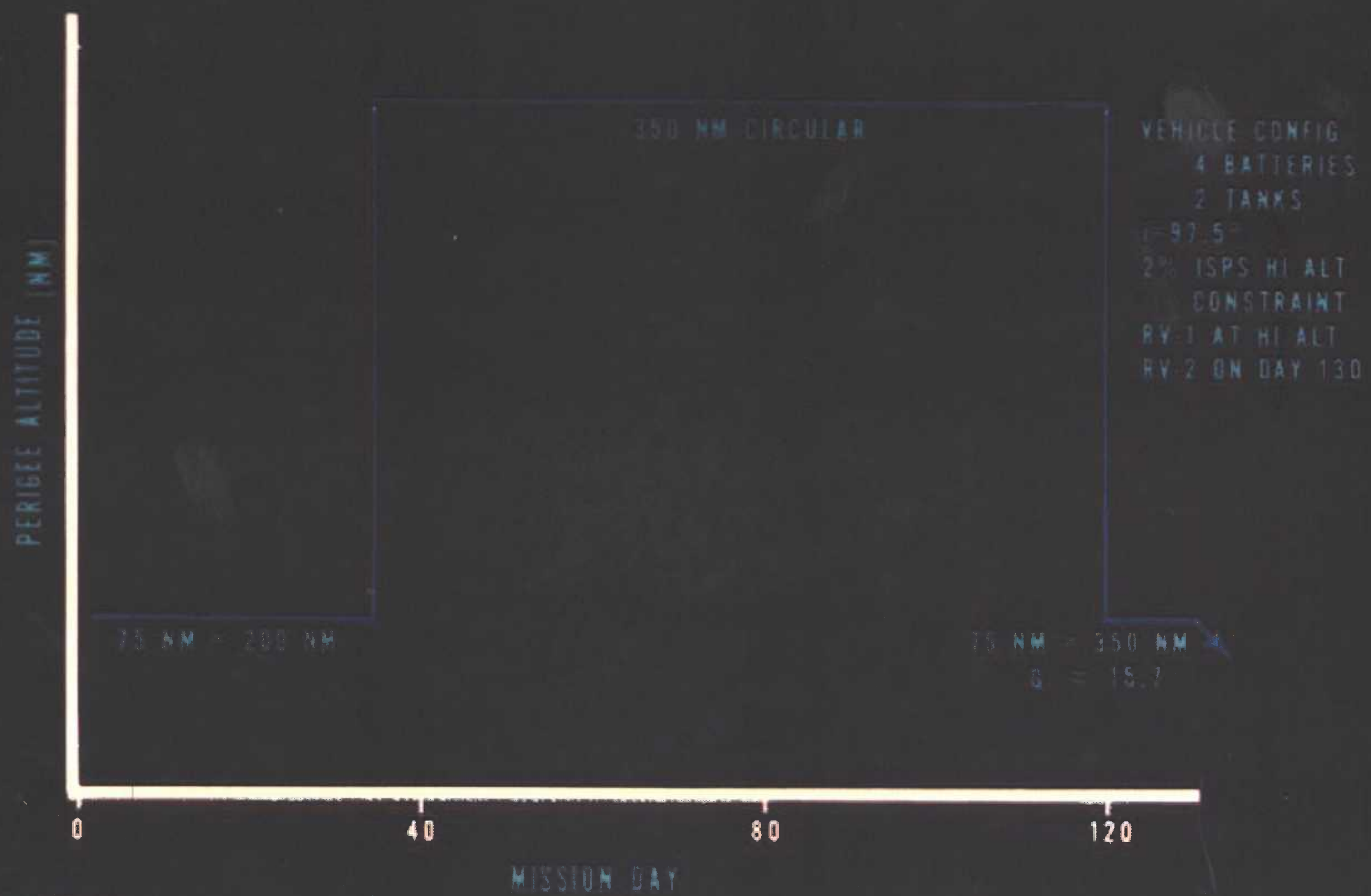


FIGURE 6-2

FREE-WALKING ORBITS

7.0 FREE-WALKING ORBITS - The free-walking orbit is an orbit which allows unrestricted precession of the argument of perigee. Normally, low perigee elliptic orbits involve high propellant expenditures to maintain the argument within a desired latitude band. Eliminating argument control reduces significantly the propellant expenditures since orbit adjust requirements are limited to short duration burns for drag make-up, orbit Q control, and the ISPS engine constraint. The result is that the altitude over the AI will cycle between the orbit extremes throughout the mission. This provides the possibility of performing "simultaneous" Dual Mode photography on the same rev. For example, search over the AI and surveillance over the southern hemisphere.

This type of orbit permits the apsidal line (perigee - apogee line) to move at its "natural" precession rate. The natural rate of precession is a function of inclination, perigee altitude, and apogee altitude. For orbits having perigee at 75 NM and apogee between 350 and 450 NM, the rate ranges from 4.0 to 3.9 degrees per day.

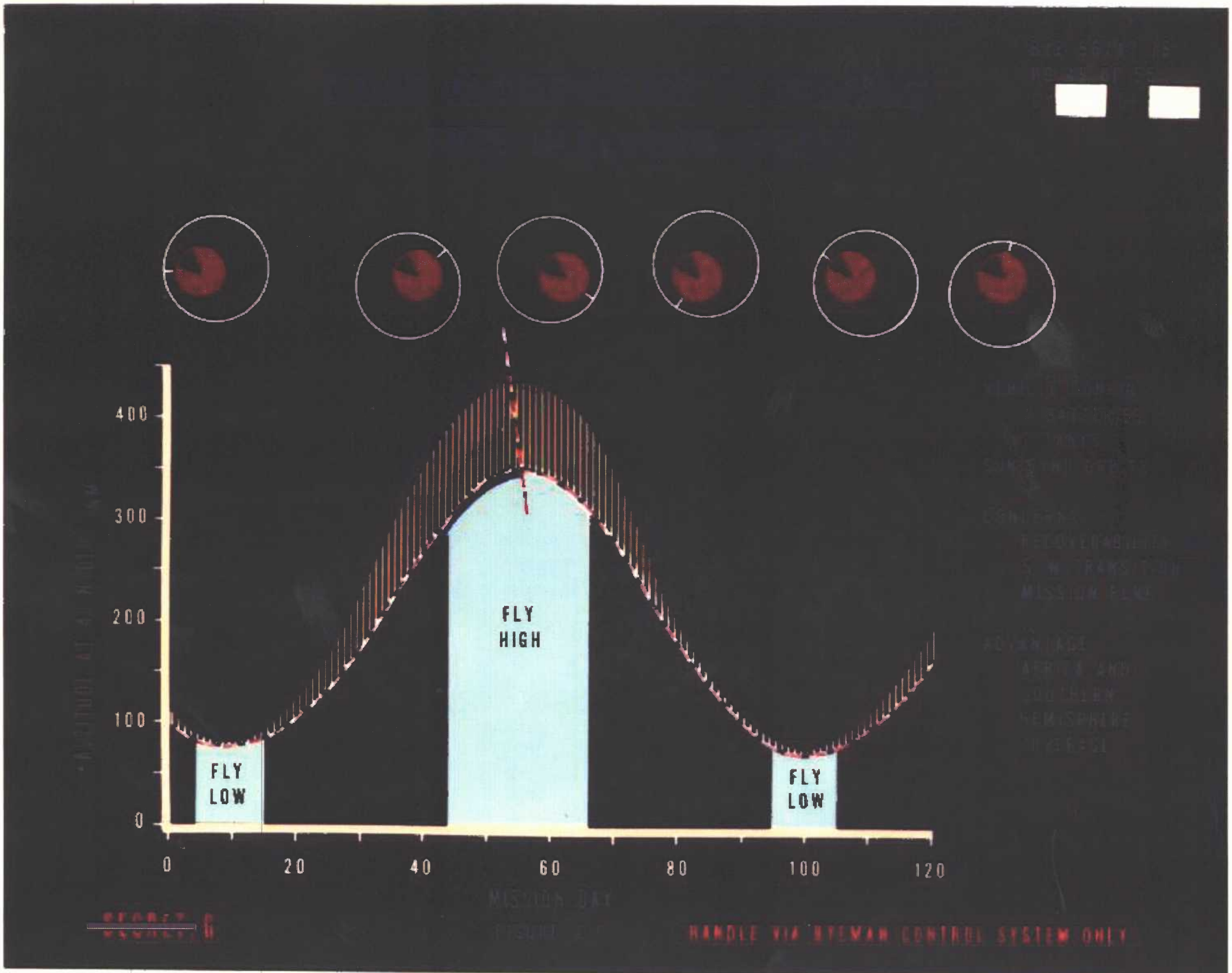
7.1 ORBIT PROFILE - The injection phase is identical to the fly high only missions both in characteristic and in the range of altitude options available. As shown in the orbit sequence in Figure 7.1, the argument of perigee begins at about the equator and continues clockwise around the world. Perigee cycles through the AI twice during a 120 day mission while apogee only once. Approximately 90 days are required to complete one apsidal revolution around the world.

Propellant expenditures are limited to drag make-up and orbit Q adjustment only. The lower perigee altitudes require drag make-up often enough that the ISPS engine constraint is satisfied. A maximum of 650 lbs of propellants, or about 40 percent of the on-orbit capacity, is required to support the mission profiles shown in Figure 7.1. These profiles were developed based upon the criteria and assumptions found in Section 3.0 with the following additions:

1) 120 mission days

2) Inject directly into desired orbit (75 NM x ? NM), i.e., 75 NM x 350 NM.

The orbit profiles presented in Figure 7.1 describe the vehicle altitude at 45° N latitude as a function of mission day. This latitude reference was chosen because it is in the center of the targeting latitude band over the AI.


400
300
200
100
0
0
20
40
60
80
100
120
FLY LOW
FLY HIGH
FLY LOW
HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

FREE-WALKING ORBITS

7.2 OBSERVATIONS - The apsidal rotation will complete 1.25 to 1.3 earth revolutions in a 120 day mission. This provides approximately forty days of optimum Dual Mode photography over the AI. The remaining eighty days are spent cycling between the altitude extremes. This does not appear attractive when considering the propellants which may be deboosted at the end of the mission.

This orbit offers the advantage of obtaining high resolution surveillance and wide area search photography of areas not normally accessed under optimum conditions. Special missions involving Africa or the southern hemisphere could be accomplished in conjunction with the normal AI mission. This is particularly attractive for longer mission duration which provides more access opportunities. A draw-back in longer missions is the delay in delivering usable products (recovery and processing) of the early target acquisitions--a subject beyond the scope of this report.

The mission duration may be extended provided the electrical power is available and possibly another hydrazine tank added. The ISPS propellant usage is not a limiting factor. The evaluation of the hardware capability to support mission extension beyond 120 days has not been completed.

Recovery from this orbit involving capsule ejection when approaching apogee will require additional study. Current recovery data does not extend to the pitch down angles and altitudes involved in this orbit.

The current mission software design planned for the Dual Mode Era will require loading the fly high algorithms and target files when transferring from the fly low orbit. The capability is not planned to provide simultaneous high and low selection algorithms and target files. Thus, message generation and software transition may become a serious impediment to this "simultaneous" Dual Mode capability. This can be overcome by changing the design concept for the Dual Mode Software Development.

RADIATION ORBIT ANALYSIS

8.0 RADIATION ORBIT ANALYSIS - The Van Allen radiation problem was discussed in Section 2.2.4. The full impact will not be fully understood until the current shielding/fogging analyses are finished. A preliminary evaluation has been completed using models of the radiation environment and vehicle shielding.

8.1 CIRCULAR ORBITS - The preliminary analysis indicates the 120 day missions at fly high altitudes up to 450 NM circular can expect up to 20 percent loss of image quality. The detailed analysis will be available at the completion of the study, however, all of the circular orbits discussed in Sections 3.0 through 6.0 appear flyable. The exact impact will require further analysis.

8.2 ELLIPTIC CONTROLLED ARGUMENT ORBITS - The primary objective of this orbit is to minimize the altitude the vehicle is at during its passage through the South Atlantic Anomaly while providing high altitude search coverage of the AI. This can be done in the following manner:

1) Inject into a 75 NM x ~200 NM orbit to maximize on-orbit propellants. This will place the argument of perigee at approximately zero degree latitude.

2) Allow argument of perigee to walk north, uncontrolled, to approximately 70^{o} N latitude. This will take about 15 - 20 days. Perform fly low photography during this segment.

3) Perform a transfer orbit to place apogee at 400 NM to 450 NM over the area of interest. Perigee will be between 150 to 250 NM.

4) Control the argument of perigee between 30^{o} and 60^{o} S latitude

8.2 ELLIPTIC CONTROLLED ARGUMENT ORBITS - Continued

Shown in Figure 8.2 is the profile of the capability to fly elliptic controlled argument orbits. This profile was developed based upon the criteria and assumptions found in Section 3.0 with the following additions:

1) Minimum of 10 second ISPS burn every six days (Corrosion constraint -satisfied by argument control burns)

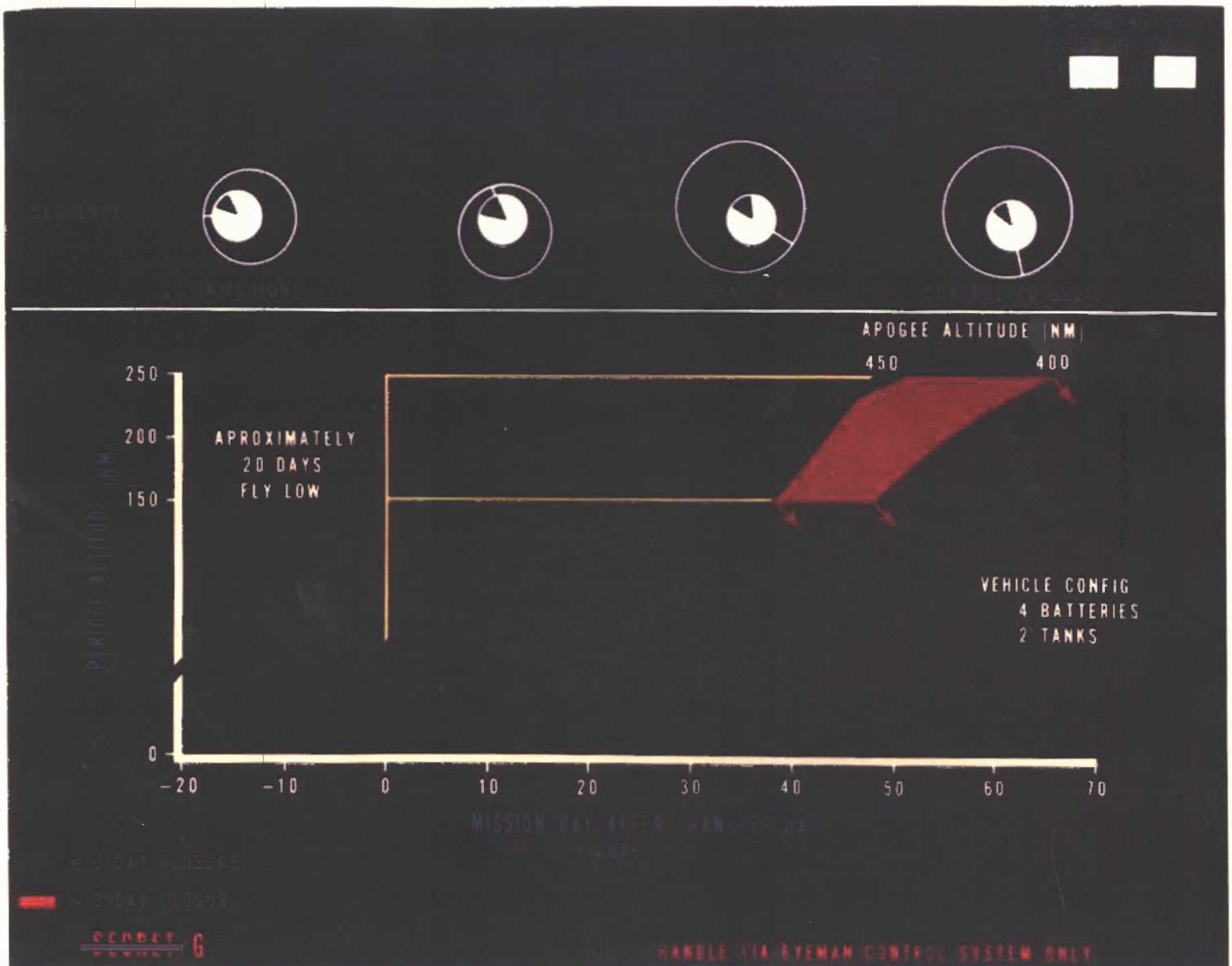
2) Sun-sync inclination

The capability of the elliptic orbit is limited in duration and coverage. Only the 250 NM by 450 NM orbit provides a reasonable, though still limited, mission duration at a desirable ($\leq$ 3 day) closure.

The elliptic orbit provides for a high daily area access due to its high altitude over the AI. However, flying longer and lower circular altitudes may provide greater total area access and will increase the opportunities for cloud-free acquisitions.

The propellant cost of making the transfer orbit is very high. These propellants could be used to expand the fly low portion of a Dual Mode mission (HI-LO or LO-HI).

The elliptic orbit is not recommended as a solution to the radiation problem, however, it is a definite capability and will continue to be investigated.


APOGEE ALTITUDE (NM)
450
400
250
200
150
0
APROXIMATELY
20 DAYS
FLY LOW
VEHICLE CONFIG
4 BATTERIES
2 TANKS
-20
-10
0
10
20
30
40
50
60
70
HANDLE VIA BYEMAN CONTROL SYSTEM ONLY

SUMMARY

9.0 SUMMARY - The orbit planning options for the Dual Mode era can be summarized as follows:

1) The options presented in Sections 3.0, 4.0, 5.0, and 6.0 are most productive (accesses) and the most efficient (ISPS). These should be considered as the basic envelope of GAMBIT Dual Mode options.

2) The "walking" orbits are not recommended due to the software incompatibility and the low percentage of optimized performance.

3) The elliptic orbits discussed in Section 8.0 are not capable of supporting intended lifetimes ($\approx$ 120 days). However, shorter lifetimes and possibly higher perigee altitudes may provide some desirable options if the radiation problem is found to limit the fly high capability. Analysis is continuing in this area.

SUMMARY

9.1 During the conduct of this analysis, it became apparent that several other potentially attractive orbital options are within the capability of the Dual Mode Vehicle. Additional investigation will be required to fully understand their potential.

9.1.1 SPECIAL ACCESS ORBITS - These orbits provide the cyclic or near cyclic (≈ even numbered Q; i.e., 16.0) access to specific areas of interest.

9.1.2 [redacted]

9.1.3 [redacted]

9.1.4 LONGER LIFE MISSION - Those orbits not limited by ISPS propellants are currently limited in power. Current studies are in progress which will extend the battery life on orbit.