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APPROACH TO RAPID MISSION DESIGN AND PLANNING

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FOREWORD

This report describes the major tasks and results accomplished during the course of the Targeting and Mission Design Study. The work was performed by Lockheed's Huntsville Research & Engineering Center while under contract to the Aero-Astrodynamics Laboratory of Marshall Space Flight Center. This work was conducted in fulfillment of Contract NAS8-26578.

CONTENTS

Section 1
INTRODUCTION AND SUMMARY

The completion of the Apollo program will cause a shift in NASA efforts to other programs such as Skylab, Application Technology Satellite program, etc. Many of these new programs will not have available the long mission planning and design lead time of the Apollo program.

The mission designer will require tools which will provide the targeting, mission planning and design with a quick reaction time for a wide variety of mission profiles, payloads, and launch vehicles. The existing complex of computer programs was developed for large, liquid-fueled launch vehicles and manned missions with long lead times. Many of these computer programs, however, can be adapted to the needs of unmanned space flight for small solid rocket launch vehicles with short mission planning cycles.

Although these missions are less complex than the lunar landing mission, the short lead time between missions and the wide variation in missions will require a sophisticated approach to the mission planning and design. The automation of the tradeoff studies of the mission profile, launch vehicle, experiments, etc., will be required. The total mission will be optimized within the payload or mission constraints to provide maximum benefit from each mission. This optimization may include addition of piggyback payloads to a launch vehicle to utilize the total launch vehicle payload capabilities, moving the launch date to take advantage of an opportunity to maximize useful data or any number of other aspects of the mission besides launch vehicle performance.

This report describes preliminary efforts to develop methods and techniques for implementation in an automated computer system which will allow rapid assessment of the parametric data, capabilities, requirements and constraints for the various types of earth orbit missions. The automated system will have capabilities to provide:

1. Mission profile/payload optimization within specified constraints
2. Systematic sensitivity and trade-off outputs
3. Mission design by planning level
4. User input control of design process, and
5. On-line graphics and man-in-the-loop design capability.

To achieve these objectives the approach used is defined by the following steps:

1. Define mission planning and design procedures and requirements
2. Develop logic flow from experiment requirements to prelaunch design
3. Analyze existing computer programs and determine additional requirements
4. Develop computer system logic flow for executing the various mission design phases, and
5. Implement the general mission design and planning computer system.

These preliminary efforts have been directed only toward Steps 1 through 4.

The mission planning and design procedures and requirements were defined using two typical missions as examples. These missions and their associated experiment package are shown in Table 1-1. While these were used to develop the design process, effort was made to generalize the process and not limit it to a few specific missions.

Table 1-1

EXAMPLE MISSIONS AND TYPICAL EXPERIMENTS

1. High Energy Astronomical Observatory Missions.
   a. Large Area X-Ray Detector
   b. Gamma-Ray Detector
   c. Gamma-Ray Telescope
   d. Cosmic-Ray Electrons
   e. Cosmic-Ray Calorimeter

2. Small Applications Technology Satellite Missions.
   a. Remote Sensing of Ocean Color
   b. Multichannel Ocean Color Sensor
   c. Carbon Monoxide Pollution
   d. Pollution and Climatology
   e. Radar Altimetry
   f. Earth Physics Magnetometer
   g. Heat Capacity Mapping Radiometer
   h. L-Band Maritime Services
   i. Radio Frequency Interview Survey

Section 2
MISSION DESIGN

2.1 THE MISSION DESIGN PROCESS

The planning and design of a mission proceeds in sequential steps or phases from initial mission concept through the actual launch to post flight mission evaluation. Each step of the design process is more detailed than previous but the number of parameters and their range of values is narrowed by analyses of the preceding.

These have been variously defined in the literature (Refs. 1 through 5) but, in general, can be defined by the level of complexity of the mathematical models (or computer programs) used in the analysis, the completeness of the mission definition and the objective of analysis phase. The first three of these phases are shown in Fig. 2.1-1.

The first phase of mission design and planning analysis uses analytical or tabular models of the mission to define possible mission concepts consisting of combinations of launch vehicles, launch sites, tracking sites, orbital altitudes and inclinations, ascent mechanics and launch dates which satisfy mission objectives (generally earth observation coverage requirements and spacecraft weight) and primary mission constraints (orbit lifetime, etc.).

Second phase analyses would be based on "3D" integrated trajectories for ascent and orbital flight, linearized error or dispersion studies, and standard models of atmosphere, thrust time histories, guidance system, navigation accuracy, etc. Mission and system definitions would include all six orbital state parameters, launch vehicle stages and other major constraints.

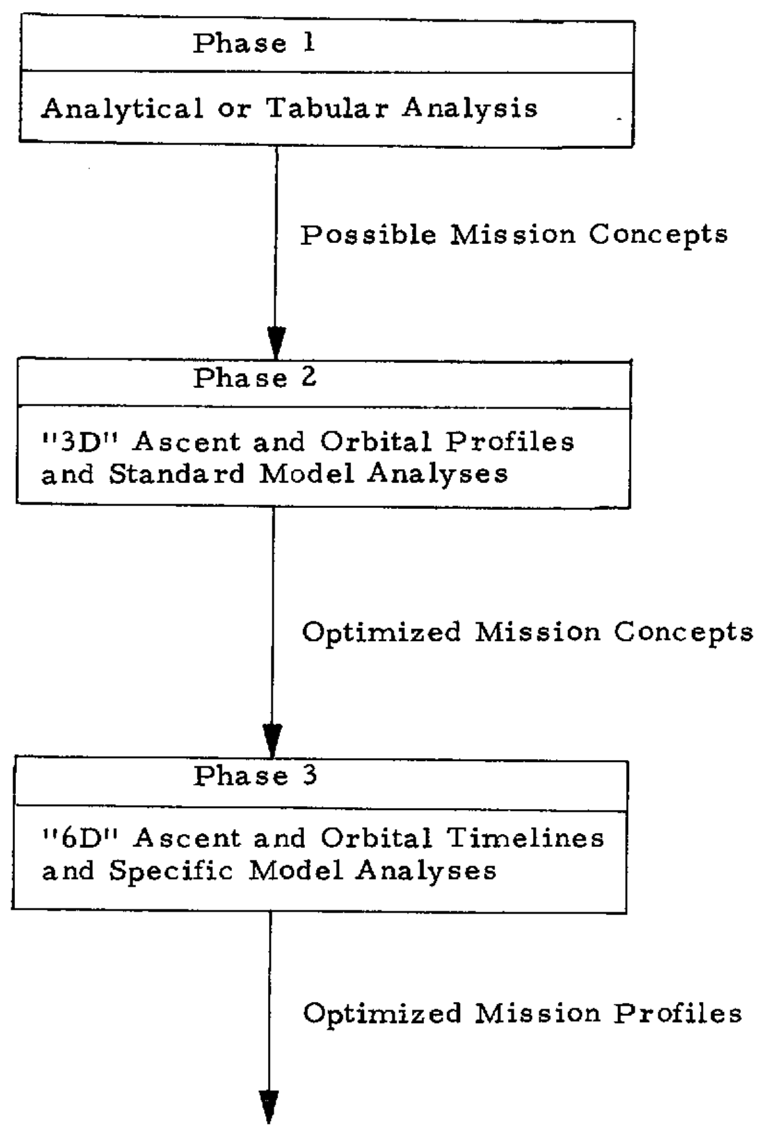

Fig. 2.1-1 - Mission Design and Planning Flow

Third phase analysis would be based upon "6D" integrated trajectories for ascent and orbital profiles, Monte Carlo error or dispersion analysis, specific models of the atmosphere, propulsion system, guidance system, etc. Mission and system definitions would include all significant mission and system constraints. The analysis objective is to optimize the mission event profile.

2.2 FIRST PHASE DESIGN

During the first phase analysis, the designer is concerned with consideration of a large number of basic mission concept alternatives, and a systematic evaluation of these alternatives is required. A recommended approach proceeding from the simplest to the most complex is:

First Pass:	Direct injection into a final circular orbit utilizing a single tracking station
Second Pass:	Adds multiple tracking station possibilities to the analysis
Third Pass:	Adds direct injection into final elliptical orbits possibilities to the analysis
Fourth Pass:	Adds indirect injection via planar parking orbits possibilities to the analysis, and
Fifth Pass:	Adds indirect injection via non-planar parking orbits possibilities to the analysis.

Within each pass, the designer would be concerned with four types of studies taken generally in sequence:

1. Parametric studies of a mission constraint as a function of its parameters
2. Design trades between mission constraints
3. Sensitivity of mission constraints to parameter perturbations, and
4. Design optimization or selection.

Three types of parametric studies would interest the designer as options:

1. Unconstrained parametric studies to indicate general characteristics
2. Parametric relationships required to satisfy a specified mission constraint value, and
3. Parametric relationships required to satisfy a specified mission constraint value but further limited to regions not violating other constraints. This option limited to consideration of a single set of launch site, launch vehicle, and tracking site analysis.

Three types of design trades between mission constraints would be available as options similar to those provided for parametric studies. Three similar options for sensitivity analysis would also be available. These options are illustrated in Table 2.2-1. A partial list of constraints is given in Table 2.2-2.

2.2.1 Parametric Studies

Table 2.2.1-1 itemizes the steps to compute a matrix of 270 lifetime computations as a function of the parameter's altitude, inclination, launch month and ballistic coefficient. This is an example of the Unconstrained Option. Three launch months are included, one giving maximum orbit lifetime during the year, one giving minimum lifetime during the year, and the earliest launch month. Three ballistic coefficient values are used, the input value, one-half, and twice the input value. Six inclination values are used, values producing maximum and minimum lifetime and values corresponding to the minimum characteristic velocity for launch from each of the four launch sites. Five values of orbit altitude are included. One altitude producing twice the desired orbit lifetimes for minimum values of the other parameters, and one altitude producing one half the desired orbit lifetime for maximum values of the other parameters. Three intermediate altitudes are also used. This option can be exercised separately or is automatically included in the other two parametric study options.

Table 2.2-1

PHASE I DESIGN OPTIONS

Designer Selects	1.	Constraints to be satisfied (list of available constraints given in Table 2.2-2)
	2.	Design passes desired (1, 2, 3 and 4)
	3.	Type of studies desired
	4.	Study options
	5.	Output desired (print, graphic, plotted, save tape)
Designer Inputs	1.	Constraint values
	2.	Ballistic coefficients, solar elevation angle, viewing angle
	3.	Earliest launch month and year
	4.	Launch vehicle group or selects launch vehicle from list
	5.	Launch site group or selects launch site from list
	6.	Tracking site group or selects tracking site from list
	7.	Observation frequency and latitude coverage desired
First Pass		Circular orbit, direct injection, single track site A. Parametric Study Type I. Unconstrained Option II. Specified Value Option III. Limited Region Option B. Design Trade-off Study Type I. Unconstrained Option II. Specified Value Option III. Limited Region Option C. Sensitivity Study Type I. Unconstrained Option II. Specified Value Option III. Limited Region Option D. Design Optimization or Selection Study Type
Second Pass		Adds Multiple Track Sites (same as First Pass)

Table 2.2-1 (Concluded)

Third Pass:	Adds Elliptical Orbits (same as First Pass)
Fourth Pass:	Adds Coplanar Parking Orbits (same as First Pass)
Fifth Pass:	Add Non-Planar Parking Orbits (same as First Pass)

Table 2.2-2

LIST OF CONSTRAINTS (FIRST PHASE)

Constraint	Parameters
Orbit Lifetime	• Ballistic Coefficient • Launch Dates • Altitudes • Inclinations • Argument of Periapsis • Circular/Elliptical
Spacecraft Weight	• Launch Vehicle • Launch Azimuth • Launch Site • Altitudes • Inclination • Direct/Indirect • Circular/Elliptical • Booster Types
Percentage of Orbit Period Solar or Stellar Body Occulted	• Orbit Altitudes • Orbit Inclinations • Orbit Nodal Position • Earth-Sun Line • Earth-Stellar Line • Ephemeral Time
Sun-Synchronous Orbit	• Altitude • Inclination
Ground Contact Opportunities Minutes per Day	• Latitude of Ground Station • Ground Stations Available • Orbit Altitudes • Orbit Inclinations • Minimum Tracking Angle
Ground Viewing Coverage	• Launch Month • Solar Elevation Angle • Altitudes • Inclinations • Viewing Angle • Observation Frequencies • Latitude Coverage

Table 2.2-2 (Concluded)

Constraint	Parameters
Cost	• Launch Site • Tracking Stations • Booster
South Atlantic Anomaly	• Altitude • Inclination • Ephemeral Time • Experiment Radiation Sensitivity

Table 2.2.1-1

ORBIT LIFETIME PARAMETRIC STUDIES – UNCONSTRAINED OPTION

1. Enter Solar Activity portion of lifetime computation routine (shown in section 2.2.5.1) and determine launch month (LM_{MAX}) producing maximum multiplier value in 12 month period following earliest launch date. Determine launch month (LM_{MIN}) producing minimum multiplier value. Select third launch month (LM_{EL}) as the earliest launch date.
2. Compute three ballistic coefficients: Input ballistic coefficient BC_{IN}, $BC_1 = \frac{BC_{IN}}{2}$, $BC_2 = 2BC_{IN}$.
3. Compute maximum altitude (h_{MAX}) of interest as altitude satisfying twice the input lifetime constraint at inclination I_{MIN}, LM_{MIN}, BC_2. I_{MIN} is built into program logic and represents the inclination producing minimum lifetime.
4. Compute minimum altitude (h_{MIN}) of interest as altitude satisfying one-half the input lifetime constraint at inclination I_{MAX}, LM_{MAX}, BC_1. I_{MAX} is built into program logic and represents the inclination producing maximum lifetime.
5. Compute five altitudes: h_{MAX}, h_{MIN}, $h_2 = \frac{h_{MAX} + h_{MIN}}{2}$, $h_3 = \frac{h_{MAX} + h_2}{2}$ and $h_4 = \frac{h_{MIN} + h_2}{2}$.
6. Compute lifetimes for all combinations of altitude (5), inclinations (6 built-in program), ballistic coefficient (3), and launch dates (3).
7. Output data according to designer option.

An example of the specified value option is given in Table 2.2.1-2. The table indicates computations necessary to determine the orbit altitudes satisfying the lifetime constraint for the various combinations of launch months, orbit inclinations and ballistic coefficients. This option is the first step in the third parametric study option.

An example of the limited region option is shown in Table 2.2.1-3. This option has two outputs. One output indicates the maximum altitude allowed by the other mission constraints, and the minimum altitude allowed by the lifetime constraint for each inclination and launch month set. The other output provides the lifetime excess over the lifetime constraint available while still satisfying the other mission constraints.

2.2.2 Design Trade-Offs

The unconstrained option computations for spacecraft weight trades is indicated in Table 2.2.2-1. Five spacecraft weight values are selected in the range of 0.25 to 4.0 times the input weight constraint. For each of the other constraints, a maximum and a minimum value are obtained providing the range of trades between spacecraft weight and the other mission constraints. In a similar manner, trades between the other constraints can be developed.

The limited region option to compute the trades available between constraints above the specified constraints levels can be accomplished by using the appropriate limiting altitudes obtained from the parametric studies — limited region option to limit the maximum and minimum values obtained in the design tradeoffs-unconstrained option (step 3 of Table 2.2.2-1).

2.2.3 Sensitivity Studies

The unconstrained option for orbit lifetime sensitivity studies (as an example) would compute the changes in orbit lifetime due to positive and negative perturbations in ballistic coefficient (10%), launch date (one month), altitude (5%), and inclination (5 degrees). The perturbations would be computed

Table 2.2.1-2

ORBIT LIFETIME PARAMETRIC STUDIES – SPECIFIED VALUE OPTION

1. Perform orbit lifetime parametric studies – unconstrained option
2. Test all 270 cases and determine whether computed lifetime is greater or smaller than lifetime constraint
3. Interpolate 5 altitude values for each ballistic coefficient, launch month, and inclination set to determine altitude value satisfying lifetime constraint, and
4. Output data according to designer option

Table 2.2.1-3

ORBIT LIFETIME PARAMETRIC STUDIES – LIMITED REGION OPTION

1. Perform orbit lifetime parametric studies – specified value option

2. Perform spacecraft weight parametric studies – specified value option

3 to N Perform mission constraint parametric studies – specified value option

N+1 Determine minimum altitude of altitude sets (step 2 to N) for each inclination value.

N+2 Output altitudes from step N+1 and step 1

N+3 Compute orbit lifetime for each launch month for altitudes for step N+1

N+4 Output lifetimes

Table 2.2.2-1

DESIGN TRADE STUDIES – UNCONSTRAINED OPTION
SPACECRAFT WEIGHT TRADES

1. Compute five spacecraft weight values:

   w_1 = input spacecraft weight

   $w_2 = 0.5\ w_1,\ w_3 = 0.25\ w_1,\ w_4 = 2w_1,\ w_5 = 4w_1$

2. Use specific value option of spacecraft weight parametric studies to compute altitudes satisfying weight constraint of w_1 value at each inclination.

3. Compute lifetime for each altitude and inclination.

4. Use peaking routine to find maximum value of lifetime for w_1.

5. Repeat Steps 2, 3, 4 for w_2, w_3, w_4, w_5.

6. Output maximum lifetime versus spacecraft weight.

7. Use peaking routine on data of Step 3 to find minimum value of lifetime for w_1.

8. Repeat Step 7 for w_2, w_3, w_4, w_5.

9. Output minimum lifetime versus spacecraft weight.

10. Repeat Steps 3 to 9 for second constraint.

11. Repeat Steps 3 to 9 for each other constraint.

12. Trades between other constraints would be developed similarly.

for each element of the matrix of lifetime values given as the output of parametric studies — orbit lifetime unconstrained option.

The specific value option (for the lifetime sensitivities as an example) would only compute the changes in orbit lifetime at the input lifetime constraint limit due to perturbations in the parameters. The lifetime limits cases would be those from specific value option parametric studies.

The limited region option (again for the lifetime example) would compute the perturbation produced changes in lifetime only for those cases defined by the parametric studies - limited region option.

2.2.4 Design Optimization/Concept Selection

The approach recommended for first phase concept selection is based upon a translation of all constraint limits into minimum characteristic velocity requirements, a grouping of mission concepts (launch sites, launch vehicles, tracking site combinations) by the amount of excess characteristic velocity obtainable, and the selection of the minimum cost combination within each group as the possible mission concept for that group. This assumes that spacecraft weight is always an input constraint, and that cost considerations are always of major concern.

Orbital energy considerations dictate a characteristic velocity requirement to achieve circular orbit as a function of altitude. The launch site azimuth restriction and latitude location determine a launch site penalty in characteristic velocity as a function of orbit inclination. This launch site penalty added to the orbital energy requirement gives the characteristic velocity requirement for achieving a given orbit altitude and inclination. Launch vehicle payload capabilities can be defined in terms of these characteristic velocities.

Each launch site has a specific inclination (I_{LS}) which provides a minimum launch site penalty. A vehicle launched from a site at I_{LS} has a minimum characteristic velocity requirement, and therefore, maximum payload at that inclination.

The various mission constraints (lifetime, communication sites, etc.) can be investigated (parametric studies - specified value option) to determine the minimum altitudes satisfying each constraint at inclination I_{LS}. The minimum altitude (for each constraint) can be converted into an orbital energy characteristic velocity and added to the launch site inclination penalty to determine minimum characteristic velocity required to satisfy mission constraints for launches from that launch site. For the lifetime constraint, minimum altitude should be selected based on the input spacecraft weight constraint and ballistic coefficient, and the launch month producing maximum lifetime. A tracking site must be specified before applying the Opportunities constraint.

If the maximum of the several minimum characteristic velocities (one for each constraint, except spacecraft weight) is chosen, this represents the minimum characteristic velocity requirement (V_c) for a launch site/tracking site combination. These minimum velocity requirements can be determined for all combinations of launch site/tracking site (say 4 launch sites and 10 tracking sites for 40 combinations).

A given launch vehicle can deliver the desired minimum spacecraft weight (input constraint value) to a specified characteristic velocity (V_{LC}). If this V_{LC} is less than the minimum V_c requirement for a given launch site/ tracking site then the mission concept represented by that launch vehicle, launch site and tracking site is unacceptable. If this V_{LC} is greater than V_c, the mission concept is acceptable and the excess characteristic velocity (ΔV_{EXCESS}) is available to trade for additional spacecraft weight, lifetime, etc., and is a measure of the mission concept's performance. The acceptability and ΔV_{EXCESS} would be computed for mission concepts. This would yield about 400 mission concepts if 10 launch vehicles are considered.

To provide meaningful performance/cost relationships it is desirable to group the mission concepts by levels of ΔV_{EXCESS}. A logical grouping of the concepts would be to search out the maximum ΔV_{EXCESS}, and the median ΔV_{EXCESS} values, and compute:

Level 1 = 0.333 median ΔV_{EXCESS}

Level 2 = 0.667 median ΔV_{EXCESS}

Level 3 = median ΔV_{EXCESS}

Level 4 = median ΔV_{EXCESS} + 0.333 (maximum ΔV_{EXCESS} - median ΔV_{EXCESS}).

Level 5 = 2 level 4

Level 6 = maximum ΔV_{EXCESS}

If the mission concept which has the lowest cost within each ΔV_{EXCESS} level is selected, then a meaningful trade-off of mission cost and mission performance is presented to the designer for his consideration and final selection of a First Pass mission concept. This mission concept selection process is summarized in Table 2.2.4-1.

2.2.5 Mission Constraint Computation Logic

This section describes some of the procedures and techniques that would be required in the initial mission design efforts. These procedures and techniques are those that are presently being used by mission designers. The constraints described in this section are a partial list of those that could be required. These constraints are those that are commonly the limiting factors on most missions.

2.2.5.1 Earth-Orbit-Lifetime Estimation

This section provides information for estimating orbital lifetime based on descriptions of initial orbit parameters, spacecraft ballistic coefficient, and launch date.

The most accurate orbital lifetime estimates are made using computer programs which integrate differential equations of motion for orbiting spacecraft in the presence of all external forces. Such programs require extensive input

Table 2.2.4-1

PHASE 1, FIRST PASS MISSION CONCEPT SELECTION FLOW

1. Compute launch site minimum V_c penalty at inclination I_{LS} for all launch sites.
2. Compute minimum altitude satisfying each mission constraint for each I_{LS} and select largest altitude (h_{max}) of each I_{LS} set. Determine (h_{max}) for all combinations of launch sites and tracking sites.
3. Compute orbital energy characteristic velocity (V_{oe}) for each h_{max}.
4. Sum V_{oe} and V_c to define minimum characteristic velocity requirement for each tracking site/launch site combination.
5. Compute characteristic velocity capability (V_{LC}) of each launch vehicle for the desired spacecraft weight.
6. Compute ΔV_{EXCESS} for each combination of launch vehicle and launch site/tracking site.
7. Compare ΔV_{EXCESS} values and group mission concepts into levels.
8. Compute minimum cost mission concept for each ΔV_{EXCESS} level.
9. Output possible mission concepts rated by cost and system performance.

data and computer execution time. The accuracy of other estimating techniques depends primarily on the assumptions made with regard to upper atmosphere density and its variation as a function of solar activity. The necessity to make predictions regarding solar activity during the whole orbit lifetime introduces uncertainties in results obtained using any estimation technique. Reference 6 presents a method for the approximate prediction of orbital lifetimes based on a time dependent atmospheric density model from which the information presented in this section has been adapted. The method was developed for a specific class of elliptical orbits with inclinations of approximately 30 degrees and eccentricities greater than zero but significantly less than unity. The method, of course, gives the best results for this class of orbits, but, for advance planning purposes, the results are generally satisfactory over a wide range of orbit parameters. The method is not defined for circular orbits since argument of perigee (undefined for circular orbits) is one of the dependent variables that must be specified. In some cases, this restriction may be circumvented by considering the fact that a truly circular orbit is an idealization and actual orbits always have some eccentricity. Moreover, because of solar radiation pressure, the atmosphere is not distributed symmetrically about the Earth. In this respect, a near-circular orbit may be assumed to have an "effective" perigee somewhere above the night-side of the Earth.

Orbit lifetime may be estimated on the basis of the following expression:

$$\text{Lifetime} = \left[\frac{M}{C_D S}\right] L_1 L_2 L_3$$

where L_1 is factor that provides effect of orbit inclination and argument of perigee on orbit lifetime

L_2 is factor that provides effect of orbital apogee and perigee altitudes on orbit lifetime

L_3 is factor that provides effect of solar activity on atmospheric density and resulting lifetime effect. Solar activity is a function of perigee altitudes and actual calendar dates.

$\frac{M}{C_d S}$ = spacecraft ballistic coefficient, kg/m^2, used as a dimensionless multiplier

M = orbiting mass, kg

S = reference area, m^2 (see table below)

C_d = drag coefficient (see table below)

This expression may be solved by an iterative process as shown in Fig. 2.2.5.1-1. The integrated $+2\sigma$ solar activities curves result in conservative lifetime estimates so that there is approximately a 95% probability that the orbit lifetime will be somewhat greater than that obtained using this procedure.

ESTIMATION OF FREE-MOLECULAR-FLOW DRAG COEFFICIENT AND REFERENCE AREAS FOR CALCULATING BALLISTIC COEFFICIENT

Satellite Orientation	Drag Coefficient, C_d	Refeɫence Area, S
Stabilized Body	2.06-2.2	Projected area
Simple Shape	2.06	Projected area
Complex shape	2.2	Projected area
Tumbling Body	2.18	1/4 total surface area

2.2.5.2 Launch Vehicle and Mission Profile Definition

After the mission designer has converted the experiment objectives into general mission requirements (i.e., orbit altitude, inclination, minimum payload, the launch site), the launch vehicle and the mission profile must be selected to

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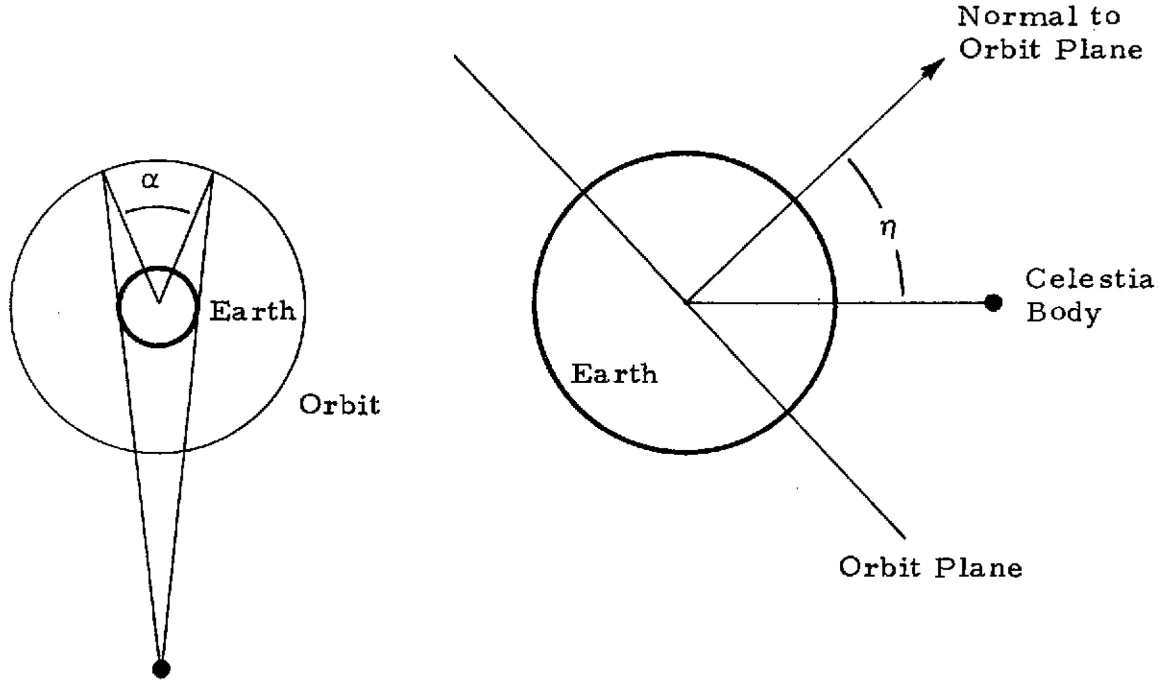
optimize the mission. To simplify the comparison the altitude inclination requirements are converted to characteristic velocity requirements. Initially the characteristic velocity required to achieve a circular orbit with a direct coplanar boost is found based on the circular orbit altitude. The characteristic velocity penalty required for the plane change to achieve the desired inclination from each of the launch sites is determined. The characteristic velocity required to achieve the circular orbit and the characteristic velocity penalty for the inclination plane change are summed to provide the V_c requirement for the direct injection circular orbit mission profile. The above procedure is repeated to obtain the characteristic velocity requirements for each of the basic mission profiles; direct injection into elliptic orbit, indirect injection into circular orbit and indirect injection with a plane change. These characteristic velocity requirements can then be compared with the capabilities of the various launch vehicles being considered to determine the payload that each launch vehicle can deliver for each of the mission profiles. This comparison will indicate the launch vehicle and mission profile which will best satisfy mission objectives. The above procedure is based upon the "Launch Vehicle Estimating Factors," NHB7100.5A, Office of Space Science and Application, Launch Vehicle and Propulsion Programs, January 1972 (Ref. 7). The logic flow of the above procedure is shown in Fig. 2.5.2.2-1.

2.2.5.3 Occultation

The occultation of a vehicle in orbit occurs when view of some celestial object from the vehicle is cut off by having some object pass between the vehicle and the celestial body. If solar panels are used for generation of electrical power, the time the Sun is occulted will be an important factor in the design of a mission and of the vehicle power usage. Vehicle attitude control may require that the vehicle use a fixed point such as Canopus or Polaris as a reference. Occultation of this fixed point will affect the design of the mission and the vehicle reaction control system.

The orientation of the proposed orbit is determined based upon orbit inclination, altitude and the launch site used. This orbit orientation with the location of the particular celestial body of interest are used to calculate the planar occultation angle, α, and the orbit normal celestial body angle, η.

These angles (see sketch below) are used with nomograms to determine the time per orbit that a celestial body is occulted. The general logic flow for this subroutine is shown in Fig. 2.2.5.3-1.



2.2.5.4 Sun-Synchronous Orbits

Since the earth is an oblate spheroid, perturbations will occur which continuously change the orbital elements, most notably a rotation of its node and argument of perigee. However, by proper orientation of the orbit, it is possible to cause the line of nodes to rotate 0.986 degrees per day so as to produce a nearly constant orbit plane orientation with respect to the sun as the earth moves in its orbit about the sun. Furthermore, by proper choice of orbit inclination and node a satellite will remain entirely in sunlight during an orbit. Therefore, it is possible to design an orbit which will remain entirely in sunlight throughout its lifetime. The relationships which govern the orbit selection for continuous exposure to the sun is as follows:

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$$\cos i_{eq} = \frac{0.986}{3\pi\, J_2} \left(\frac{P}{R_e}\right)^2 = \frac{0.986}{3\pi\, J_2} \left(\frac{a(1-e^2)}{R_e}\right)^2$$

where i_{eq} is the inclination of the orbit to the earth equator and is between 90^o and 180^o (i.e., retrograde). Thus, by orienting the orbit so that its node is initially 90^o from the earth-sun line and trading inclination and altitude through the use of the above equation, continuously sunlit orbits can be designed. This equation is plotted in Fig. 2.2.5.4-1.

2.2.5.5 Ground Station Contact Time

In the initial pass for designing a mission to satisfy a set of experiment objectives, the average ground contact time for an orbit will be used to determine the orbit's suitability. This suitability from the tracking standpoint will be — can the vehicle be located with the desired accuracy within the average tracking time? To satisfy the communication requirement the average ground station contact time must be long enough to allow the experiment data to be returned before satellite data storage capability is exceeded. The average ground station contact time will affect the total program cost due to the number of locations of ground stations required to satisfy communications and tracking requirements.

This segment will provide the average ground station contact time from tabular data as a function of orbit altitude, inclination and the various ground stations available. The average contact time will be generated for each available station individually for both circular and elliptic orbits. The average contact time for various combinations of ground stations will be determined using the overlap time in which more than one station has contact at the same time. The average ground station contact time is output for circular and elliptic orbits based on both the use of a single ground station and the use of multiple ground stations. The general flow logic described above is shown in Fig. 2.2.5.5-1.

EARTH ORBIT MISSION FACTORS

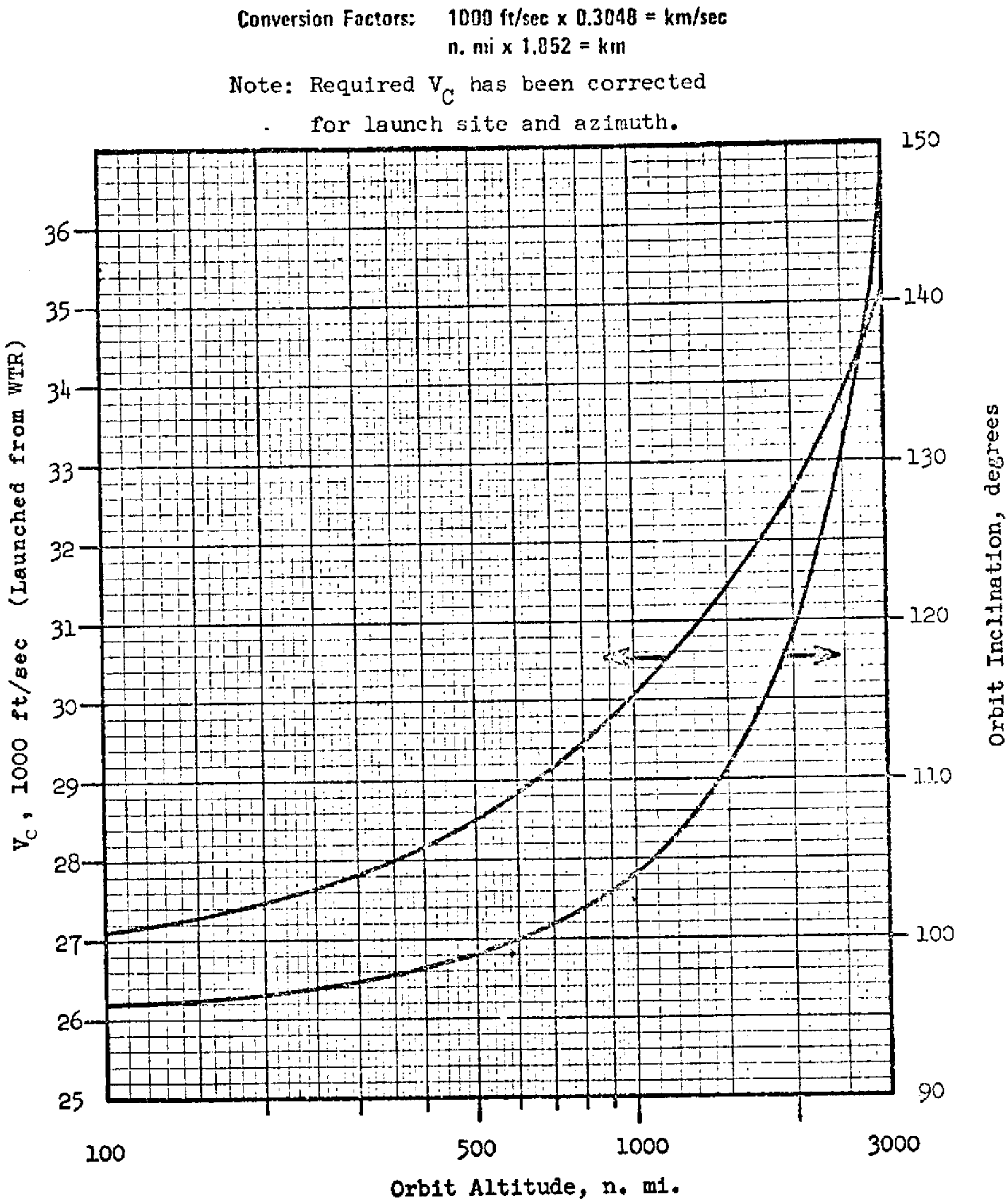

Fig. 2.2.5.4-1 - Inclination and V_c of Circular Sun Synchronous Orbits

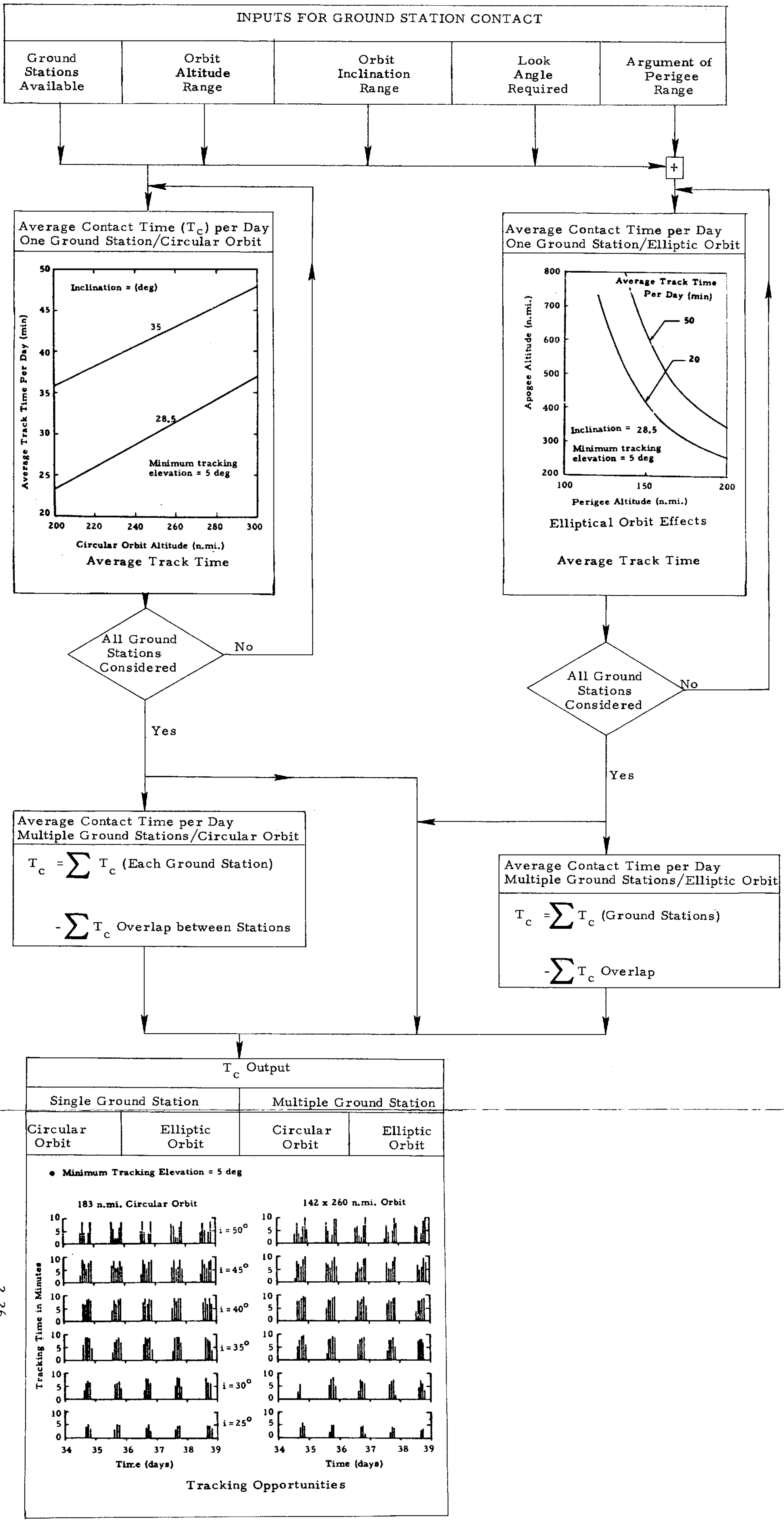

Fig. 2.2.5.5-1 - Ground Station Contact Time Determination Logic

2.2.5.6 Experiment Constraints

The mission designer must take inputs for a proposed mission experiment package and transform these into mission objectives and requirements. Figure 2.2.5.6-1 shows this transformation flow for a typical experiment that requires viewing a portion of the earth's surface. The experiment equipment capabilities (sensor focal length, film resolution, etc.) and the parameter to be measured (desired surface resolution) will determine the maximum acceptable orbit altitude. The inclination of the orbit is determined by the location of the region to be viewed and by the lighting requirements. The ground coverage angle is determined by the orbital altitude and the minimum viewing angle of the experiment sensors. This ground coverage angle with the field of view of the experiment sensors is used to find the ground swath width. The surface displacement of the ground track is calculated based on orbital altitude and inclination. This orbital track displacement is combined with the ground swath width to determine the nominal mission duration for ideal conditions. The experiment package sensitivity to radiation may require that it be shut down during passage through the South Atlantic anomaly. This shutdown time may require that the mission length be extended beyond the nominal mission length. The probability that the surface will be obscured by clouds or other effects is determined based upon the region being viewed and the season of the year. This probability is used to find the mission duration required to satisfy the mission objective. The maximum orbit altitude, orbit inclination, mission length, ground swath and the "best" launch period are passed on to the next program module. The computational technique above is a modification of the technique described in Ref. 1.

2.2.5.7 Trapped Radiation Interference

The most serious radiation problem encountered by the satellite is in the South Atlantic Anomaly. A cross section of the earth's Van Allen radiation belts (shown in Fig. 2.2.5.7-1b) shows the reason for the anomaly; i.e., an asymmetry of the geomagnetic field with respect to the geographic axis. Not only is the magnetic axis inclined to the polar axis, but it is also displaced at the center of the earth by approximately 184 n.mi. This displacement causes

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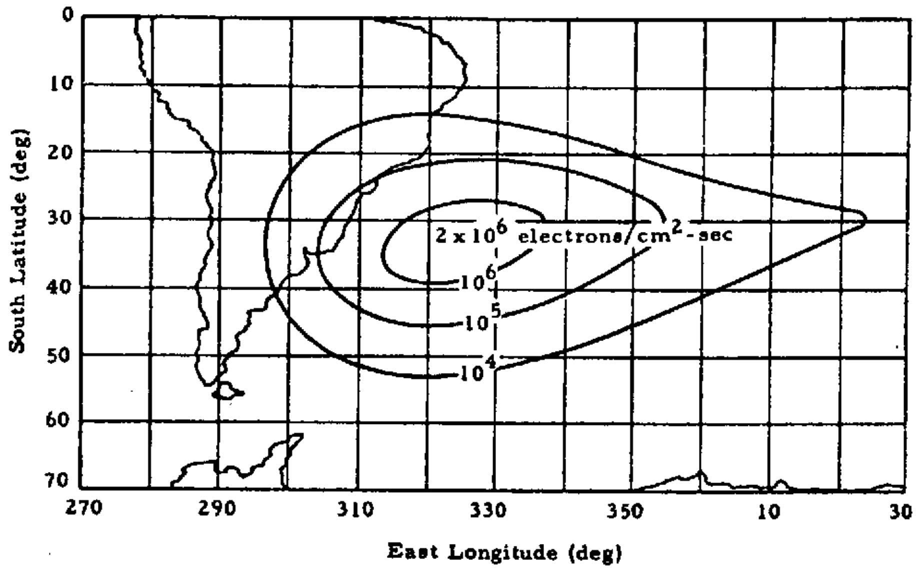
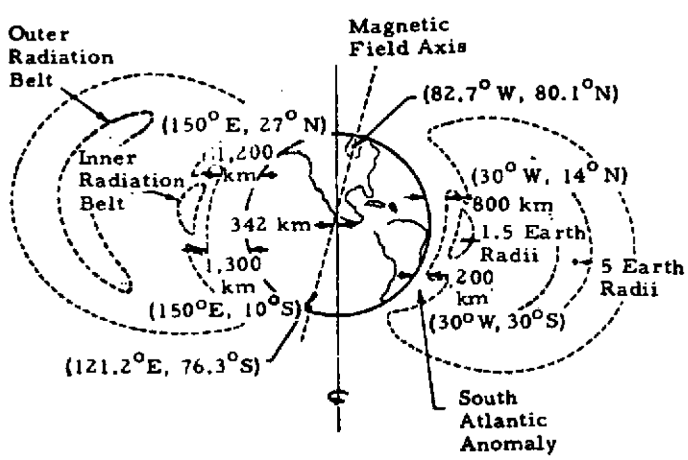
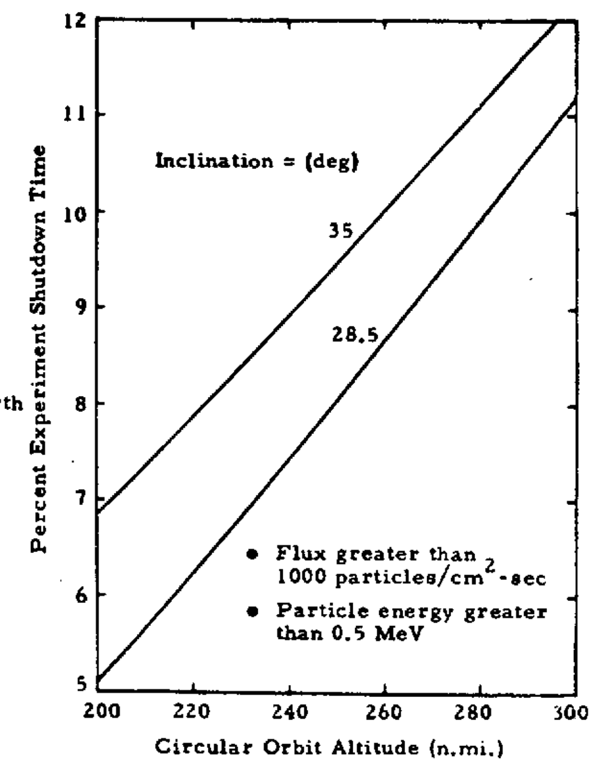


(a)



Exaggerated outlines of radiation belts, Magnetic field axis approximates location of axis of an offset magnetic dipole, but is asymmetrical. Cross section of the radiation belts is in the plane of the 30°W geographical meridian and illustrates relative altitudes at which the South Atlantic Anomaly particles mirror in that plane.

(b)



(c)

Fig. 2.2.5.7-1 - South Atlantic Anomaly

the inner radiation belt to be displaced toward the earth's center in the South Atlantic at 30 deg West longitude and 30 deg South latitude. In this area, which is called the South Atlantic Anomaly, the inner belt dips to within 108 n.mi. of the earth's surface. Figure 2.2.5.7-1a shows the electron flux counting rate in the South Atlantic Anomaly in 1966 at an altitude of 216 n.mi. At this altitude the electron flux counting rate exceeds 10^6 electrons/cm^2-sec. The proton flux intensity is about three orders of magnitude less in this region. The intensity of the radiation and the size of the contaminated area increases with altitude. The proton flux at 200 n.mi. altitude is fairly stable, but the electron flux varies over two orders of magnitude with time. One of the greatest factors influencing electron flux at low altitudes is the effect of thermonuclear (Starfish) explosions in the upper atmosphere. These detonations produce energetic electrons in the energy range which affect particle counting rates for years. Also, variations of electron flux occur in orders of magnitude on a monthly basis. Much of the variation in radiation intensity can be attributed to the day-to-day variation in solar activity. The effect on the mission is a possible loss of data when the spacecraft is in regions of high intensity radiation. Figure 2.2.5.7-1c shows the percentage of time that data obtained from the vehicle would not be relevant and experiment shutdown would occur due to saturation based on the assumption that saturation occurs at a radiation flux rate of 1000 particles/cm^2-sec and an energy level of 0.5 MeV.

Percentage shutdown time would be presented in tabulated data form. The tabulated data would have to be generated by running the CODE III program to compute shutdown times for 200 to 400 orbital revolutions for each data point. The tabulated data would be parameterized on inclination, perigee altitude and apogee altitude.

2.2.5.8 Cost

In order to design a mission within a general cost constraint and to get the most for the money spent, the mission designer will have to compare costs for various missions. For the first pass level the cost items that will be considered are launch vehicle, launch site and tracking costs. The proposed mission experiment profile is used to develop a mission time line and a mission success probability. The mission success probability is combined with the tracking, launch vehicle, and launch site costs to find an expected mission cost. This expected mission cost then is input to the mission designed to be used to determine which proposed mission has the "lowest" cost. Figure 2.2.5.8-1 shows the expected mission cost logic flow.

2.3 SECOND PHASE DESIGN

During the second phase of analysis, the designer is concerned with a limited number of basic mission concepts which must be defined and evaluated to a much higher degree of detail than accomplished during first phase analysis. A recommended approach would be to analyze each concept separately, and perform the comparative evaluation as a final step:

First Pass:	Analyze first mission concept
Second Pass:	Analyse second mission concept
	⋮
N^{th} Pass:	Analyze N^{th} mission concept
$(N+1)^{th}$ Pass:	Relative evaluation of N mission concepts

Within each of the first N passes, the designer would be concerned with four types of studies taken generally in sequence:

1. Parametric studies of a mission constraint as a function of its parameters (Table 2.3-1 provides a list of possible constraints and their parameters)
2. Design trades between mission constraints
3. Sensitivity of mission constraints to parameter perturbations, and
4. Design optimization.

The designer would be able to choose the passes and the study type as options in the program. These options would allow the designer to make an initial

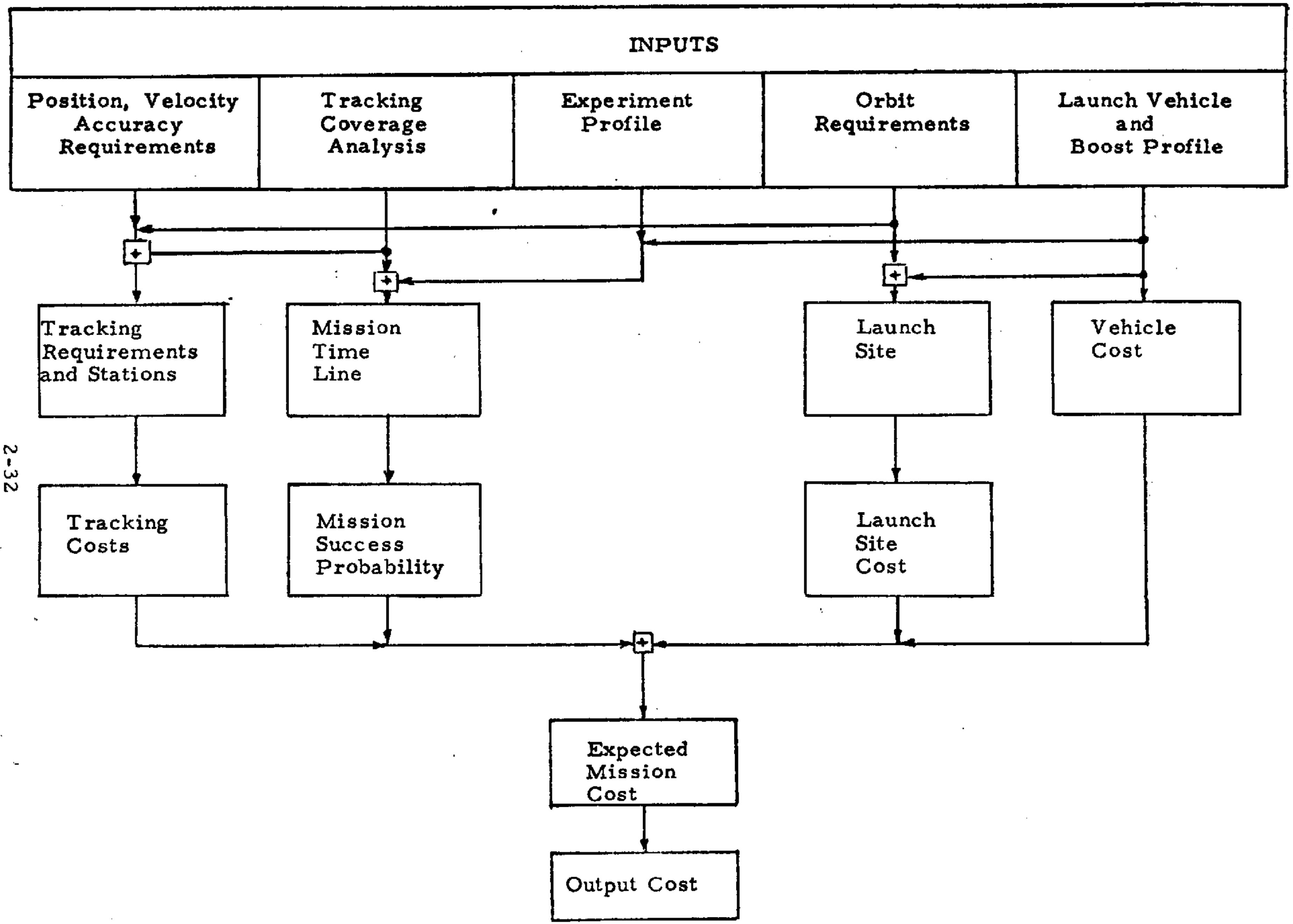


Fig. 2.2.5.8-1 - Expected Mission Cost Logic Flow

Table 2.3-1
LIST OF SECOND PHASE CONSTRAINTS

Constraint	Parameters
Orbit Decay	• All orbital parameters • Nominal attitude profile • Ballistic coefficient • Launch date
Spacecraft Weight	• Launch vehicle • Launch site latitude and longitude • Launch azimuth • Staging times • Weight drops • Thrust and specific impulse • Lift and drag coefficients • Thrust attitude profiles • Propellant loads • Vehicle weights • Orbital cutoff conditions
Stage Impact Dispersions	• All spacecraft weight parameters • Off-nominal launch vehicle parameters • All guidance parameters
Injection Accuracy	• All guidance system parameters • All spacecraft weight parameters
Tracking Accuracy	• All orbital parameters • Tracking site latitude and longitude • Minimum tracking angle • All tracking system uncertainties
Communication Opportunities per Orbit	• All orbital parameters • Minimum tracking angle • Tracking site latitude and longitude
Ground Coverage per Orbit	• All orbital parameters • Viewing angle • Ground target latitude and longitude • Observation frequencies

Table 2.3-1 (Concluded)

Constraint	Parameters
Ground Lighting per Orbit	• All orbital parameters • Ephemeral time • Launch hour • Solar elevation angle
Spacecraft Solar/Celestial Occultation	• All orbital parameters • Ephemeral time • Launch hour • Right ascension and declination of celestial body

data setup covering all possibilities, but he could restrict analysis to only the most favored concepts. The concepts analyzed could be expanded at any time by just changing the option cards. These design options and the input to the program are listed in Table 2.3-2.

2.3.1 Parametric Studies

For each mission constraint selected by the designer, computations will be made for a matrix of 20 altitude and inclination combinations. Four inclination values (built in the program) are used spanning the unrestricted launch azimuths for the given launch site and, including the launch azimuth, producing maximum load capability. Five altitude values are used. One altitude will be the maximum altitude which satisfies all the constraints (from Phase I, Design Optimization Analysis). One altitude will be the minimum altitude which satisfies all the mission constraints (from Phase I, Design Optimization Analysis) and three intermediate values.

Since spacecraft weight is always a mission constraint, the parametric studies are started by developing optimum ascent trajectories for each element of the altitude/inclination matrix. The achieved weight and the cutoff state vector will be output for use in the orbital decay parametric study and an orbital profile generator. Also output will be the impact locations of the various booster stages. Any parameter of the launch vehicle (e.g., first stage burn time) input by the designer will also be used to provide a matrix of ascent trajectories for these altitude-inclinations, and for increases and decreases of this input parameter.

In addition to these parametric variations, launch date for orbit decay studies and launch time for occultation and solar lighting will be required.

2.3.2 Design Tradeoffs and Sensitivity Analyses

The parametric study data will be used to develop design tradeoff relationships between mission constraints in a similar manner as discussed for First Phase Design.

Table 2.3-2

PHASE II DESIGN OPTIONS

Input:	Mission concepts Altitudes Inclinations Ballistic coefficient Nominal attitude profile Spacecraft viewing angle Observation frequencies Ground target latitude, longitude Identify celestial bodies from list Non-standard parameter values
Designer Selects:	1. Constraints to be satisfied (from list given in Table 2.3-1) 2. Non-standard parameters to be used (all parameters not input are built-in as standard or are optimized). Taken from list. 3. Additional parameters to be included in parametric, tradeoffs, sensitivity or design selection studies. Taken from list. 4. Output desired (print, graphic, plotted, savetape) 5. Type of study desired a. parametric b. design tradeoffs c. sensitivities d. design optimization 6. Passes (mission concepts) desired for study.

Sensitivities of the mission constraints to parameter perturbations will be computed for the matrix of altitude/inclination combinations and the other parameters varied in the parametric studies.

2.3.3 Design Optimization

The approach recommended for Second Phase Design Optimization is based upon evaluating each altitude-inclination combination in terms of satisfying the various mission constraints and the express performance margins available, and selecting the design with the highest performance margins.

The first step is to test the spacecraft weights, communication opportunities, etc., constraints computed in the parametric study option to determine if these constraints are all satisfied. The second step is to determine if all constraints are satisfied in the presence of expected parameter deviations.

An example of this second step for the spacecraft weight constraint is to determine the degradation of spacecraft weight (ΔW) expected from deviations in launch vehicle parameters from nominal values. Thus to satisfy the spacecraft weight constraint in step two, the launch vehicle would have to deliver the desired spacecraft weight (W_s) plus the expected weight loss to the altitude-inclination orbit.

The third step would be to determine the excess performance margin available for that orbit. Referring to the spacecraft weight example, the excess performance margin in spacecraft weight (W_{EPM}) is:

$$W_{EPM} = W_c - (W_s + \Delta W)$$

where W_c is the computed spacecraft weight delivered to that orbit.

The fourth step would be the selection of the orbit with the largest performance margin. Since the performance margins could be a vector of spacecraft weight, communication times per orbit, etc., a selection criteria for relative weighting of these margins could be input by the designer.

2.3.4 $(N+1)^{th}$ Pass: Concept Selection

Each of the N mission concepts input by the designer will have been evaluated and optimized at this point in terms of performance margins. Selection of a concept from this group will also require input of a criteria by the designer.

2.3.5 Mission Constraint Computation Programs

The procedures, techniques, and computer programs for performing the basic constraint computations involved in Second Phase Design are available in operational form at Lockheed and MSFC. These programs are briefly described below.

1. <u>MSFC/LMSC Orbital Analysis Program</u>

   Brakensiek, R. R., "User's Manual for the MSFC/LMSC Orbital Analysis Program," LMSC-HREC D162647, Lockheed Missiles & Space Co., Huntsville, Ala., March 1971.

2. <u>ROBOT</u>

   Gottlieb, R. G., and S. J. Johnson, "ROBOT - Apollo and AAP Preliminary Mission Profile Optimization Program," Report AAR-102, Applied Analysis, Inc., Huntsville, Ala., April 1968.

3. <u>CODE III</u>

   Johnson, B. C., "Coast, Orbit, Deboost, Entry III (CODE III) Program User's Manual," LMSC-HREC D225815, Lockheed Missiles & Space Co., Huntsville, Ala., June 1972.

4. <u>Guidance System Error Program</u>

   Staggs, W. H., and J. A. Stanton, "Guidance System Error Program User's Manual," LMSC-HREC A711007, Lockheed Missiles & Space Co., Huntsville, Ala., April 1965.

5. <u>Tracking Accuracy Program</u>

   Bissett-Berman Corporation, "Final Report on Study of Orbit Error Analysis - Contract No. NAS8-21109," September 1967.

6. Orbital Maneuvers (QUOTA)

McDaniel, G. A., "User's Manual for the Quasi-Optimal Trajectory Applications Program," LMSC-HREC D162030, Lockheed Missiles & Space Co., Huntsville, Ala., December 1969.

2.4 THIRD PHASE DESIGN

During the third phase of analysis, the designer is concerned with a single mission concept which must be defined and evaluated in great detail. The design objective is to optimize the mission event sequence. The great detail associated with the mission definition implies a large number of parameters. Most of these are fixed or can be selected from system design or physical considerations, so that a relative few need be considered in the design process. However, the ones that are needed vary with the specific mission concept being studied. The approach recommended for this phase of design is to define standard input and output formats for the various analysis program, develop routines for handling data transmissions between them, and let the designer set up his own design sequence.

Section 3
CONCLUSIONS AND RECOMMENDATIONS

With the completion of the Apollo program and the shift of NASA efforts, a tool is needed by the mission designer to provide the quick reaction time required for the wide variety of mission profiles, payloads, and launch vehicles. Although the techniques are presently available to design a mission, the designer must do a large portion of the work "by hand." This report has outlined several steps in the mission design procedure to develop an approach to rapid mission design and planning. The development of an automated mission design system appears to be both possible and desirable. The designs must be "man-in-the-loop" but by automating certain design steps the time required to complete the design can be reduced.

Efforts should be directed to further definition of the design procedures and reducing these procedures to a form which can be programmed. The capability to combine several of the available design programs to provide the design data in the required form, should be developed.

Section 4
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3. Wolverton, R. W., Flight Performance Handbook for Orbital Operations, John Wiley and Sons, Inc., New York, 1963.

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6. Haussler, J.B., and E.D. Geissler, "A Graphical Method for the Prediction of Satellite Lifetime Based on a Time-Dependent Atmosphere Density Model," George C. Marshall Space Flight Center, Orbit Mechanics Branch, NASA, September 1971.

7. NASA, "Launch Vehicle Estimating Factors for Advanced Mission Planning," NHB 7100.5A, 1972 Edition, NASA Headquarters, Office of the Director of Launch Vehicle and Propulsion Program.