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EGO LAUNCH WINDOW STUDY

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EGO LAUNCH WINDOW STUDY

I. INTRODUCTION

The purpose of this study is to determine whether a launch time exists, such that the practical restraints, imposed by the spacecraft and experiments, can be satisfied. This report deals primarily with the spacecraft imposed restraints. The experiment imposed restraints will be considered in a later report.

II NOMENCLATURE

a = Semimajor axis of the elliptical orbit ~ kilometers

e = Eccentricity of the orbit

i = Inclination = The angle between the orbit plane and the equatorial plane ~ degrees

Ω = Longitude of ascending node -- the angular distance from the vernal equinox measured eastward in the equatorial plane to the point of intersection of the orbit plane where the satellite crosses from south to north ~ degrees

Line of Apsides - The line connecting perigee and apogee

ω = argument of perigee = the angular distance measured in the orbital plane from the line of nodes to the line of apsides ~ degrees

h_p = Perigee height = this is defined as the distance from the surface of the reference sphere (of radius equal to the Earth's equational radius) to the spacecraft measured along the radius vector when the spacecraft is nearest to the reference sphere ~ kilometers

Φ = Geocentric latitude = this is defined as the angle at the center of the Earth between the radius through a given point and the equatorial plane ~ degrees

θ = Terrestrial longitude = the angular distance from the Greenwich meridian, measured eastward along the equator to the meridian plane ~ degrees

h = Geocentric height = this is defined as the distance from the reference sphere (of radius equal to the Earth's equatorial radius) to the spacecraft measured along the radius vector

v_s = The speed of the vehicle ~ ft/sec

A_c = Azimuth = the compass heading measured clockwise from north ~ degrees

E = Elevation Angle = the angle between the velocity vector and the plane normal to the radius vector passing through the vehicle ~ degrees

III. ASSUMPTIONS *

A. Initial Conditions

The nominal geocentric polar coordinates (See Section II for definition) at injection with their $\pm 3\sigma$ tolerance are as follows:

$$\Phi = -13.927 \quad \pm \quad 0.284 \text{ degrees}$$

$$\theta = 124.372 \quad \pm \quad 0.492 \text{ degrees}$$

$$h = 282878.0 \quad \pm \quad 14636.0 \text{ meters}$$

$$v_s = 10641.0 \quad \pm \quad 14.0 \text{ meters per second}$$

$$A_c = 61.932 \quad \pm \quad 0.323 \text{ degrees}$$

$$E = 1.29126 \quad \pm \quad 0.098 \text{ degrees}$$

B. Launch Window Restraints

1. The perigee height will stay above its initial value for a one (1) year period. This eliminates the possibility of the satellite losing its energy, due to aerodynamic drag, and plunging into the Earth.

2. The maximum eclipse time per orbit shall not exceed two (2) hours for a one (1) year period. For eclipse times of a longer duration the temperature of the solar array would go below an allowable limit.

3. At least 262 minutes will elapse between injection and the first eclipse. There will be a drain on the battery power supply from lift-off to acquisition of attitude. In order for the batteries to have sufficient energy to sustain the first eclipse, it will be necessary for the spacecraft to spend at least 262 minutes in sunlight between injection and the first eclipse.

4. The injection point will not lie in darkness. The attitude control system is not designed to acquire the spacecraft in darkness.

* Mr. R. E. Russey of the OGO Project Office furnished these initial conditions and restraints.

C. Mathematical Model

1. Perturbing forces included in the model -- The perturbing forces due to the Sun, the Moon, and the gravitation field of the Earth (2nd, 3rd and 4th harmonic) were included in the analysis for the construction of the "Launch Window" map. (Figure 1).

2. Perturbing forces not included in the model -- The perturbing forces due to the aerodynamic drag and the radiation pressure were not included in the analysis for the construction of the "Launch Window" map. (Figure 1).

Figures 2 through 8 show that radiation pressure and aerodynamic drag have a negligible effect on the EGO orbit for a launch time of July 15, 1963 at 4 hour UT. Each figure presents three (3) curves, two of which were determined with the Interplanetary Trajectory Program, and the third with the Launch Window Program. The difference between the two curves generated with the Interplanetary Trajectory Program is that one includes the effects of aerodynamic drag and radiation pressure and the other does not. The third curve, the one generated with the Launch Window Program is included on the figures to show the agreement between the programs.

The aerodynamic drag coefficient is defined in the conventional manner

$$C_D = \frac{D}{\frac{1}{2} \rho U^2 A}$$

where U is the flight speed of the spacecraft relative to the atmosphere of density ρ and A is the cross-sectional area of the body normal to the flight direction. A realistic value of $C_D A = 170$ square feet was assumed. The value of $C_D = 2.5 \pm 0.5$ is recommended by Schamberg (Reference 1). The cross sectional area A for EGO varies between 10 square feet and 90 square feet.

The acceleration due to solar radiation is (I/c) K (A/M), where I is the solar energy flux, C is the velocity of light and K is a constant ($0 \leqq K \leqq 2$) whose value depends on the reflecting characteristics of the surface. (Reference 2). For a satellite which is transparent to the solar flux I, K = 0. For specular reflection from a flat plate, K = 2. The cross-sectional area A which is normal to the vector from the Sun to the satellite varies from 10 square feet to 90 square feet and the mass of the spacecraft is about 1000 pounds. The solar flux I = 2.00 cal/cm^2 - minute, K = 1 and (A/M) = 0.17409 square centimeters per gram was assumed for this analysis or equivalently a perturbing acceleration of about 8.1×10^{-6} $\frac{cm}{sec^2}$.

Figure 9 presents another comparison between the Interplanetary Trajectory Program and the Launch Window Program for a launch date of July 15, 1963, at 0.4 hours UT. For this launch date the perigee is lower and aerodynamic drag should have a larger effect. This comparison lends further credence to the assumption of zero aerodynamic drag and zero radiation pressure used in the Launch Window Program.

IV. METHOD OF ANALYSIS

Two IBM 7090 computer programs were used to generate the results of this report. A discussion of the two programs follows:

A. The Launch Window Program

This program was used to establish that part of Figure 1 corresponding to the restraints: i.e., perigee shall not go below its initial value for a one (1) year period and the maximum eclipse time per orbit shall not exceed two (2) hours for a one (1) year period.

The input quantities to this program are the orbit elements a_o, e_o, i_o, Ω_o and ω_o at the initial time τ_o. The program calculates a, e, i, Ω and ω as a function of time τ, influenced by the gravitational forces due to the Moon, the Sun and the Earth. The time per orbit that the satellite spends in the Earth's shadow is also in the program's output. In other words, the program calculates the shape of the orbit (a, e) and the orientation of the orbit in space (i, Ω and ω) as a function of time τ, but it does not calculate the position the spacecraft has in the orbit (true anomaly, etc.)

With regard to perigee, the perigee altitude is calculated for a time when the spacecraft is not in general at perigee. This may be done because perigee height changes gradually over one period of the satellite's orbit.

The time that the satellite spends in shadow is based on a set of elements (a, e, i, Ω, and ω) and the Sun's position vector which are for a time τ. In general the spacecraft is at some arbitrary point at time τ and not necessarily in shadow at that time. It is assumed that the orbital elements and the Sun's position vary so slowly over one period of the satellite's orbit that the shadow time per orbit can be calculated by using the aforementioned elements, the Sun's positions and Kepler's law.

The gravitational effects of the Moon are computed using Halphen's method as modified by P. Musen to facilitate the numerical computation. (Reference 3). This method permits the numerical integration of the long range lunar effects over an interval of many years. The short period terms are removed from the disturbing function by averaging it over the mean anomalies of the Moon's orbit and the satellite's orbit. Comparison between the curves calculated with this program and the Interplanetary Trajectory Program, (Figures 2, 5, 7 and 9) show that this program averages out the periodic Moon terms (with a period of 28 days), whereas the Interplanetary Trajectory Program includes the short periodic Moon terms.

The gravitational effects of the Sun are computed by means of a series expansion of the disturbing function. (Reference 4). The short period terms (of period equal to the satellite's period) are removed by averaging the disturbing function over the mean anamoly of the satellite's orbit. The periodic terms (the terms with a period of 182½ days or the time required for the Earth to revolve half way around the Sun) are included in the calculations by this program. This is evidenced by Figures 2, 5, 7, 9, 10 and 12.

The Earth's perturbing gravitational potential is represented by the second, third and fourth zonal harmonics (page 34 of Reference 5).

The effects of periodic terms are not included since for the purpose of this work only the secular effects are needed. The shadow computation was developed by A. J. Smith of the Theoretical Division. A. J. Smith also did the programming of the basic orbital computations including Halphen's method. R. Devaney and H. Montgomery of the Theoretical Division adapted the basic program to OGO problems.

The advantage of this program is its speed. It takes about $4\frac{1}{4}$ minutes of computer time to compute a one year orbit using a five (5) day integration interval.

B. <u>The Interplanetary Trajectory Program</u>, (Reference 6)

This program was used to establish that part of the "Launch Window" map (Figure 1) which corresponds to the restraint: at least 262 minutes shall elapse between injection and the first eclipse, and the injection point will not be in darkness.

This program was also compared with the shadow calculation of the Launch Window Program for mutual verification.

The input quantities to this program are the geocentric polar coordinates, ϕ, θ, h, v_s, A_c, E, at the initial time. (Options exist to input the initial conditions in other coordinate systems).

A variety of outputs are generated but only those outputs of interest to OGO will be mentioned. The program calculates the osculating elements a, e, i, ω, Ω, and the mean anomaly M corresponding to any subsequent time τ, influenced by effects of additional bodies, the 2nd, 3rd and 4th zonal harmonics (these are functions of the satellite's declination) and the 2nd tesseral harmonics (these are functions of the right ascension of the satellite with respect to Greenwich) of the Earth, aerodynamic drag, triaxial lunar potential, thrust, and radiation pressure. The times per orbit that the satellite spends in the refracted Earth's shadow (umbra and penumbra) are also included in the program's output.

This is a versatile program but requires correspondingly greater amounts of computer time. It takes about 1 to 1½ hours of computer time to compute a one year EGO type orbit.

V. RESULTS AND DISCUSSION

A. Based on Nominal Injection Conditions

Figure 1 presents a map of all of the possible launch days and injection hours from May 30, 1963 to July 3, 1964. Only those launch times corresponding to the areas labeled "all restraints satisfied" will satisfy the four restraints innumerated in Section III. Those areas labeled "eclipse exceeds 2 hours per orbit T days after launch" satisfy restraints 1, 3 and 4 for a whole year but satisfy restraint 2 only for T days. For example, August 18, 1963 at 0 hours U.T., satisfies all restraints for 330 days as labeled on the map. All other areas on the map are excluded.

Figure 1 shows that July 15, 1963 at 4 hours U.T., is a completely acceptable launch date. The data presented in figures 2 thru 8 are for this launch date. These figures show good agreement between the two computer programs and that aerodynamic drag and radiation pressure may be ignored. For this launch date, Figures 2 through 8 are as follows:

FIGURE	ORDINATE	ABSCISSA
2	Perigee height, H_p	Days after launch
3	Eclipse time	Days after launch
4	Argument of perigee, ω	Days after launch
5	Eccentricity, e	Days after launch
6	Semi-major axis, a	Days after launch
7	Inclination angle, i	Days after launch
8	Longitude of the ascending node, Ω	Days after launch

Figure 2 shows that the perigee altitude goes from 280 km to 3000 km in 360 days for this launch date. The periodic Moon term (of period of 28 days) shown in the two Interplanetary Trajectory Program curves (but averaged out in the Launch Window Program) of Figure 2 has a half amplitude of about 30 km. This should be considered when interpreting the results of the Launch Window Program if perigee is allowed to go below its initial value where aerodynamic drag becomes appreciable.

Figure 3 presents eclipse time per orbit versus days after launch as calculated by the Launch Window Program and the Interplanetary Trajectory Program. This figure shows that eclipses occur for about 30 days duration with a maximum eclipse time of 1.8 hours per orbit and elipses also occur for 117 days duration with a maximum eclipse time of 0.44 hours per orbit.

Figure 4 shows that the argument of perigee ω changes from 329.5 degrees to 360 degrees in 360 days.

Figure 5 shows that the eccentricity e changes from 0.8915 to about 0.8475 in 360 days for this launch date.

Figure 6 presents semi-major axis versus days after launch for this launch date.

Figure 7 shows that the inclination changes from about 31.07 degrees to 41.0 degrees in 360 days.

Figure 8 shows that the longitude of the ascending node Ω changes from 141 degrees to 118 degrees in 360 days.

Figure 9 presents perigee altitude versus days after launch for a point on the "Launch Window" map which lies on the lower perigee boundary, i.e., a launch time of 0.4 hours UT on July 15, 1963. This point was deliberately chosen on the boundary as a realistically worst case to show the effect of neglecting aerodynamic drag (the aerodynamic drag is higher at lower altitudes).

Figures 10 and 11 present perigee height versus days after launch and shadow time per orbit versus days after launch for a marginally acceptable launch date. Figure 10 shows that the perigee restraint is met but Figure 11 shows that a two hour eclipse occurs 320 days after launch which violates the "2 hour eclipse restraint".

In order to demonstrate the sensitivity of the perigee boundary, a point was considered which is not too far removed from the boundary. Figure 1 shows that July 15, 1963 at 0 hours UT is unacceptable, but is very near the perigee boundary.

Figure 12 shows that perigee altitude becomes unacceptable after only 78 days for this launch date. This demonstrates that the perigee boundary is very sensitive with respect to launch time.

B. <u>Injection Tolerance Study</u>

Table I presents the effects of the 3 σ injection tolerances on the boundaries of the "Launch Window" map for a launch date of July 15, 1963.

Columns 3 and 4 of Table I are labeled "lower perigee boundary" and "upper perigee boundary" respectively. The "low perigee boundary" is the earliest launch hour (UT) of July 15, 1963 for which the perigee restraint enumerated in paragraph B of Section III is satisfied. The "upper perigee boundary" is the latest launch hour for which the aforementioned perigee restraint is satisfied. Columns 5 and 6 of Table I are labeled "lower shadow boundary" and "upper shadow boundary"; these columns give the earliest and latest times respectively, for which the 2 hour eclipse restraint per paragraph B of Section III is satisfied. The last row of Table I defines the boundaries for the nominal injection conditions. A comparison of this row with the others show that the 3σ tolerance affect the launch window boundary by at most 1/4 hour.

VI. CONFIDENCE IN RESULTS

The confidence in this analysis is established by comparing the results of the Launch Window Program with the results of the Interplanetary Trajectory Program and with experimental data.

A. Comparison of Computer Programs with Each Other

The two programs are based on different theories and developed completely independently. Figures 2 through 9 show that the two programs are in good agreement. Further discussion is given under paragraph C of Section III.

B. Comparison of Computer Results with Experimental Data

Figure 13 presents variation of perigee height for the Echo Satellite. The observed variation, indicated by the circles, is compared with the computed variation for a period of more than 380 days. This single computation carried out for 380 days shows remarkable agreement for a numerical integration program over such an extended number of orbits. This comparison shows that accurate orbital data are generated with these two computer programs. Therefore, there is a high degree of confidence in the data presented in this report.

VII. REFERENCES

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3. Musen, P., On the Long Period Luni-Solar Effect in the Motion of an Artificial Satellite, NASA TN D-1041, July, 1961, Goddard Space Flight Center, Greenbelt, Maryland.

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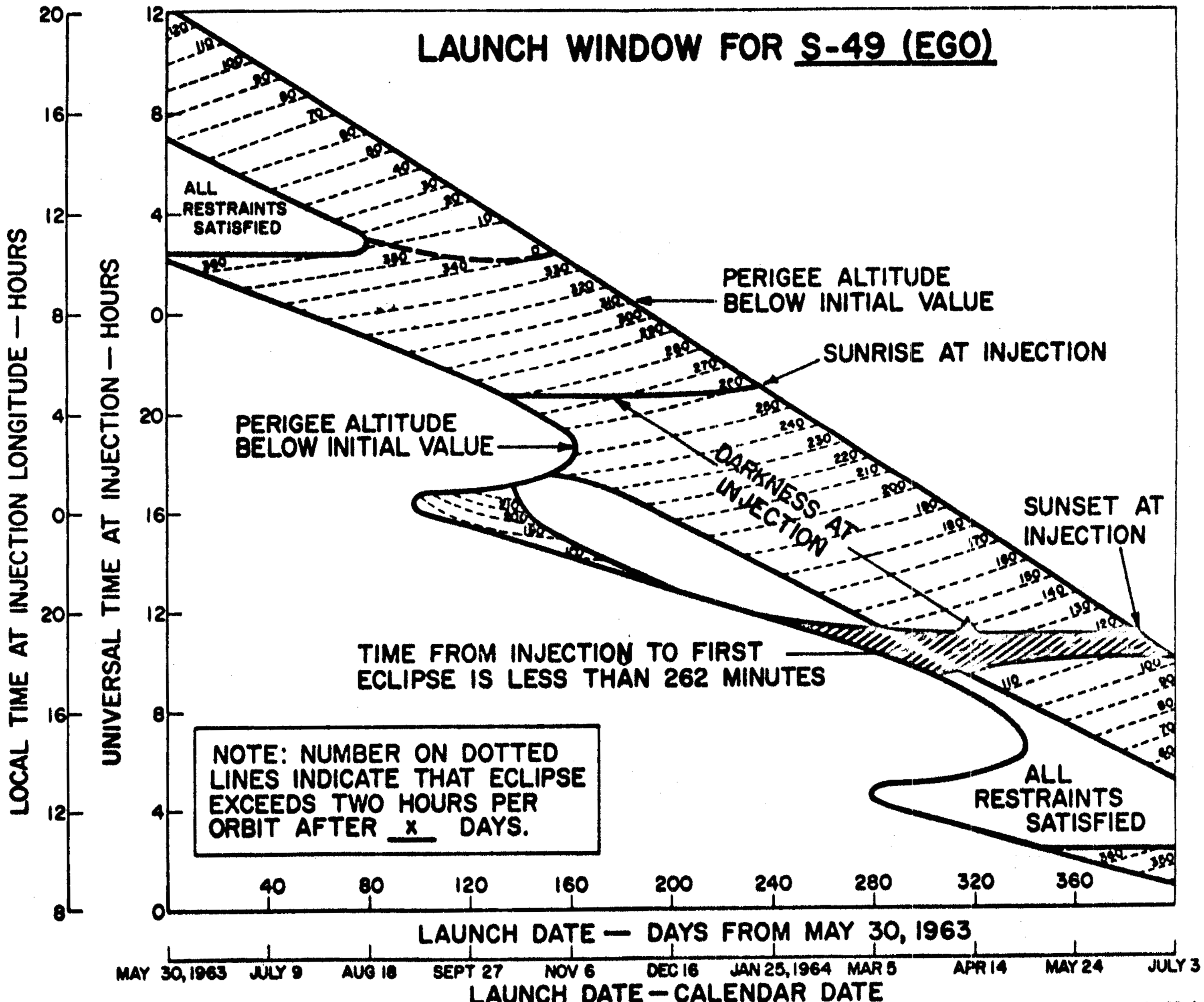
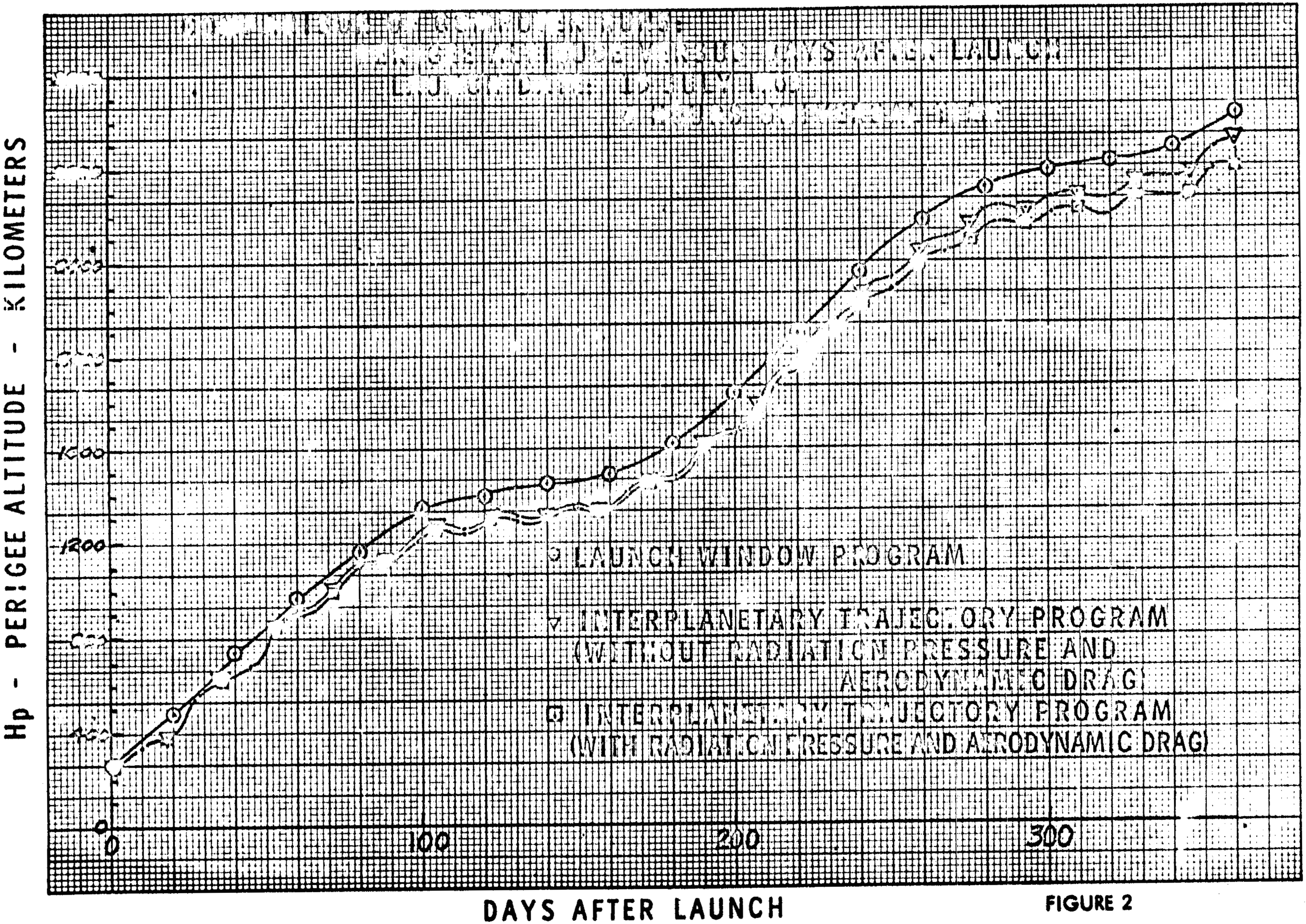
5. Kaula, W. M., Celestial Geodesy, NASA TN-D-1155, March 1962, Goddard Space Flight Center, Greenbelt, Maryland.

6. Pines, S., and Wolf, H., Interplanetary Trajectory by the Encke Method Programmed for the IBM 7090, Report No. RAC-656-451, 12 December, 1960, Research and Development Division, Republic Aviation Corporation, Farmingdale, Long Island, New York.


LAUNCH WINDOW FOR S-49 (EGO)
ALL RESTRAINTS SATISFIED
PERIGEE ALTITUDE BELOW INITIAL VALUE
SUNRISE AT INJECTION
PERIGEE ALTITUDE BELOW INITIAL VALUE
DARKNESS AT INJECTION
SUNSET AT INJECTION
TIME FROM INJECTION TO FIRST ECLIPSE IS LESS THAN 262 MINUTES
ALL RESTRAINTS SATISFIED
NOTE: NUMBER ON DOTTED LINES INDICATE THAT ECLIPSE EXCEEDS TWO HOURS PER ORBIT AFTER x DAYS.
LOCAL TIME AT INJECTION LONGITUDE — HOURS
UNIVERSAL TIME AT INJECTION — HOURS
LAUNCH DATE — DAYS FROM MAY 30, 1963
MAY 30, 1963
JULY 9
AUG 18
SEPT 27
NOV 6
DEC 16
JAN 25, 1964
MAR 5
APR 14
MAY 24
JULY 3
LAUNCH DATE—CALENDAR DATE
FIGURE 1



LAUNCH WINDOW PROGRAM
INTERPLANETARY TRAJECTORY PROGRAM
(WITHOUT RADIATION PRESSURE AND
AERODYNAMIC DRAG)
INTERPLANETARY TRAJECTORY PROGRAM
(WITH RADIATION PRESSURE AND AERODYNAMIC DRAG)
1200
1600
0
100
200
300
Hp - PERIGEE ALTITUDE - KILOMETERS
DAYS AFTER LAUNCH

FIGURE 2

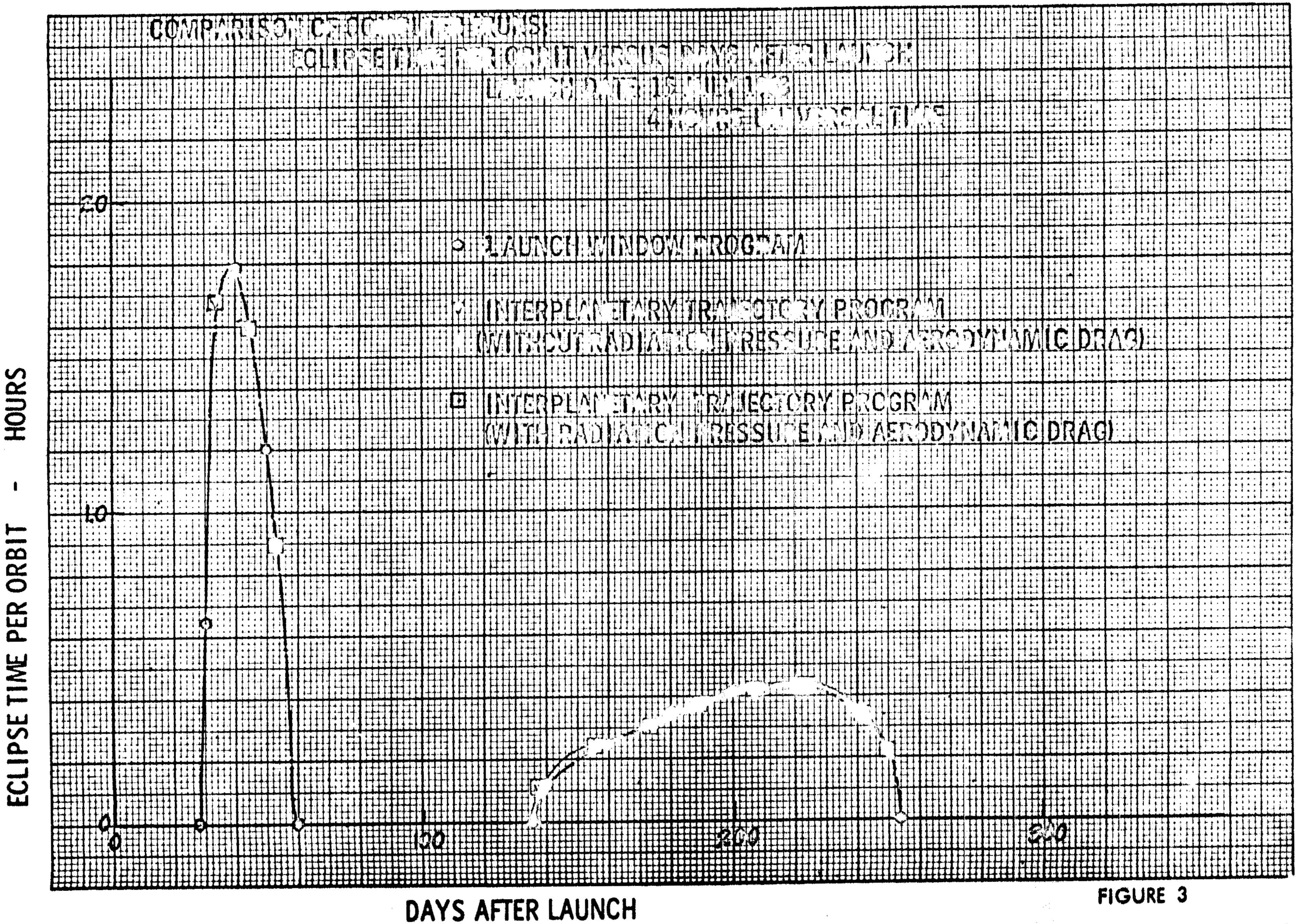
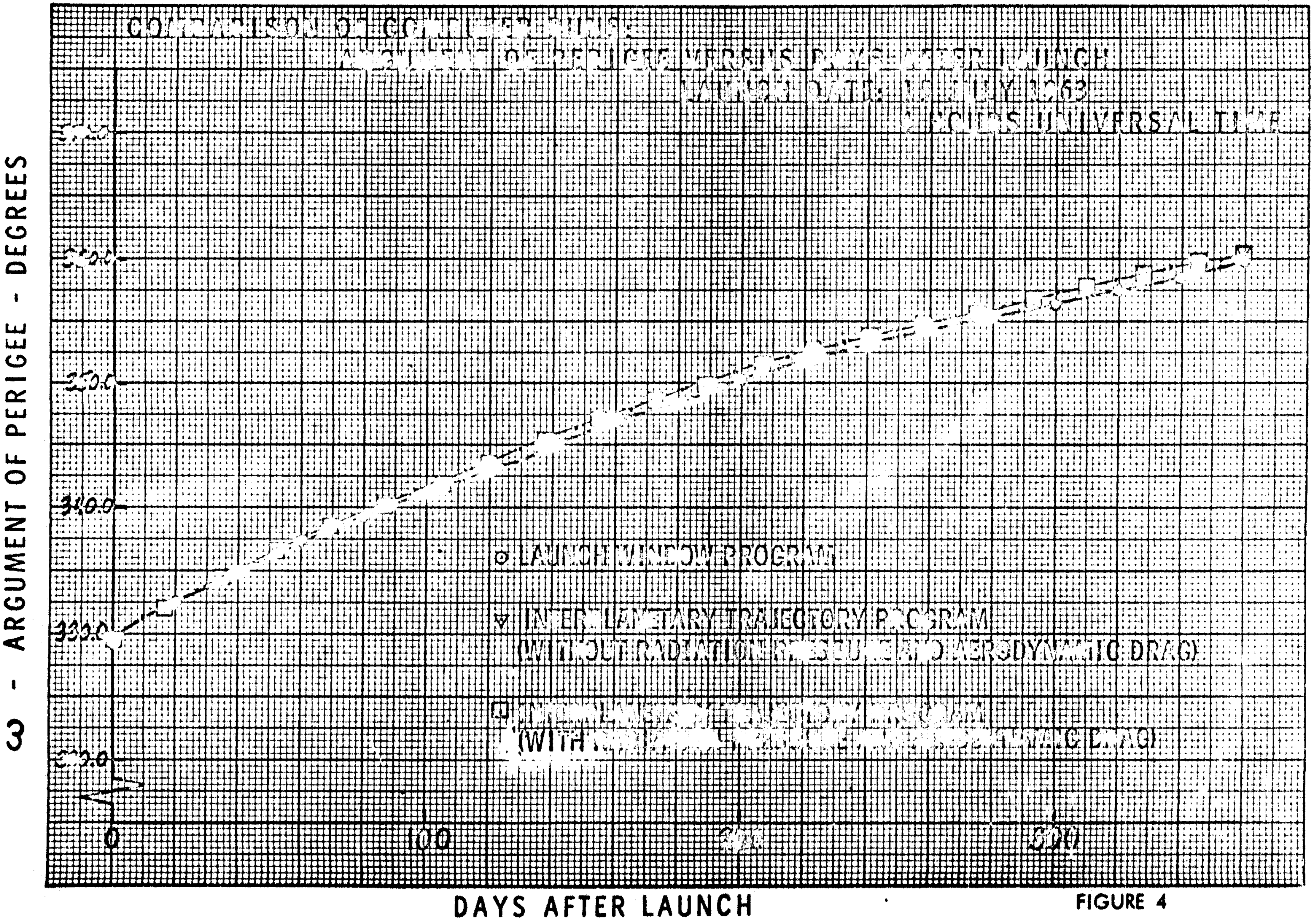
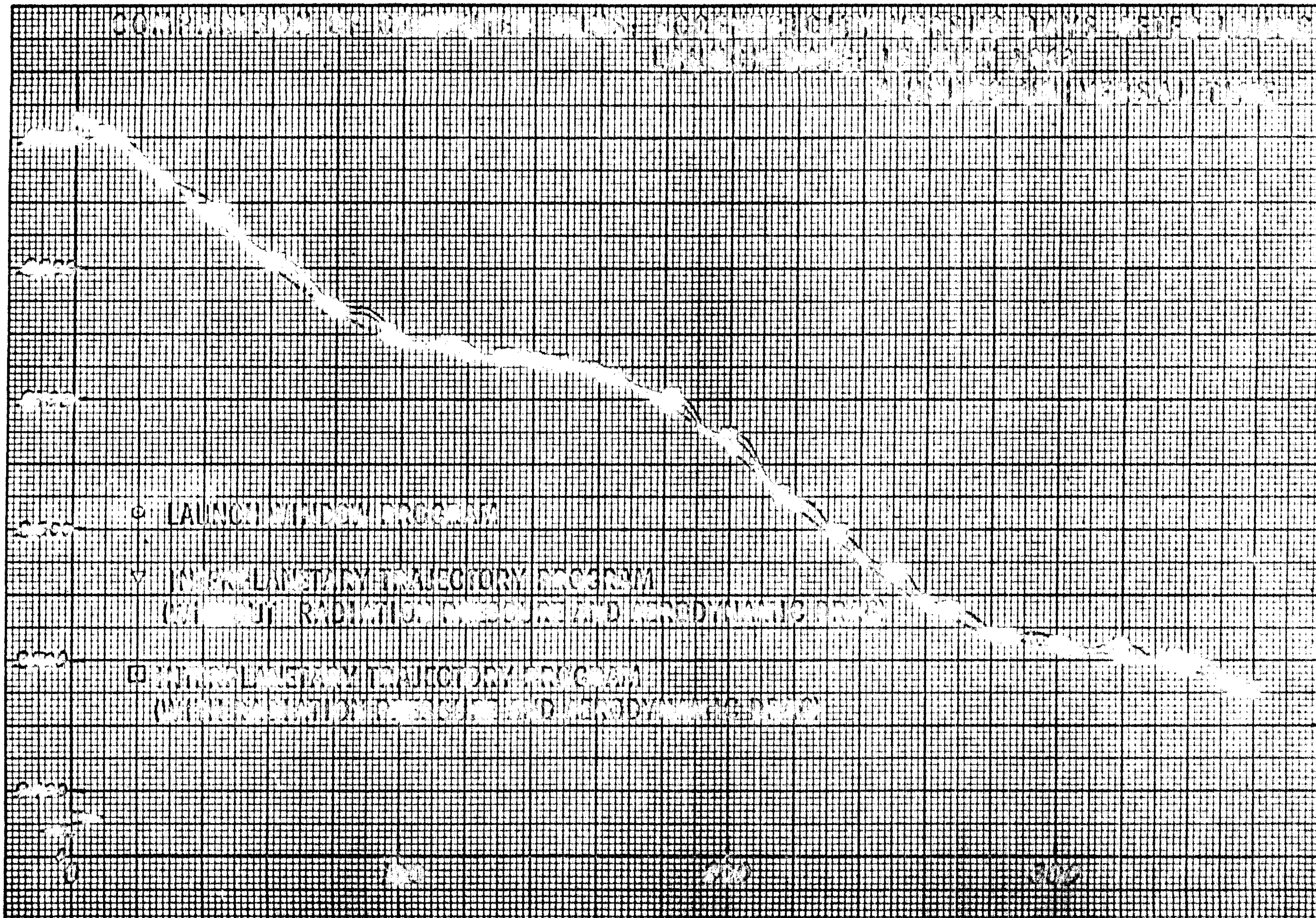

ECLIPSE TIME PER ORBIT - HOURS
20
10
0
DAYS AFTER LAUNCH
0
100
200
300
INTERPLANETARY TRAJECTORY PROGRAM
INTERPLANETARY TRAJECTORY PROGRAM

FIGURE 3



ARGUMENT OF PERIGEE - DEGREES
LAUNCH WINDOW PROGRAM
0
100
DAYS AFTER LAUNCH

FIGURE 4



e - ECCENTRICITY (NO UNITS)

DAYS AFTER LAUNCH

FIGURE 5

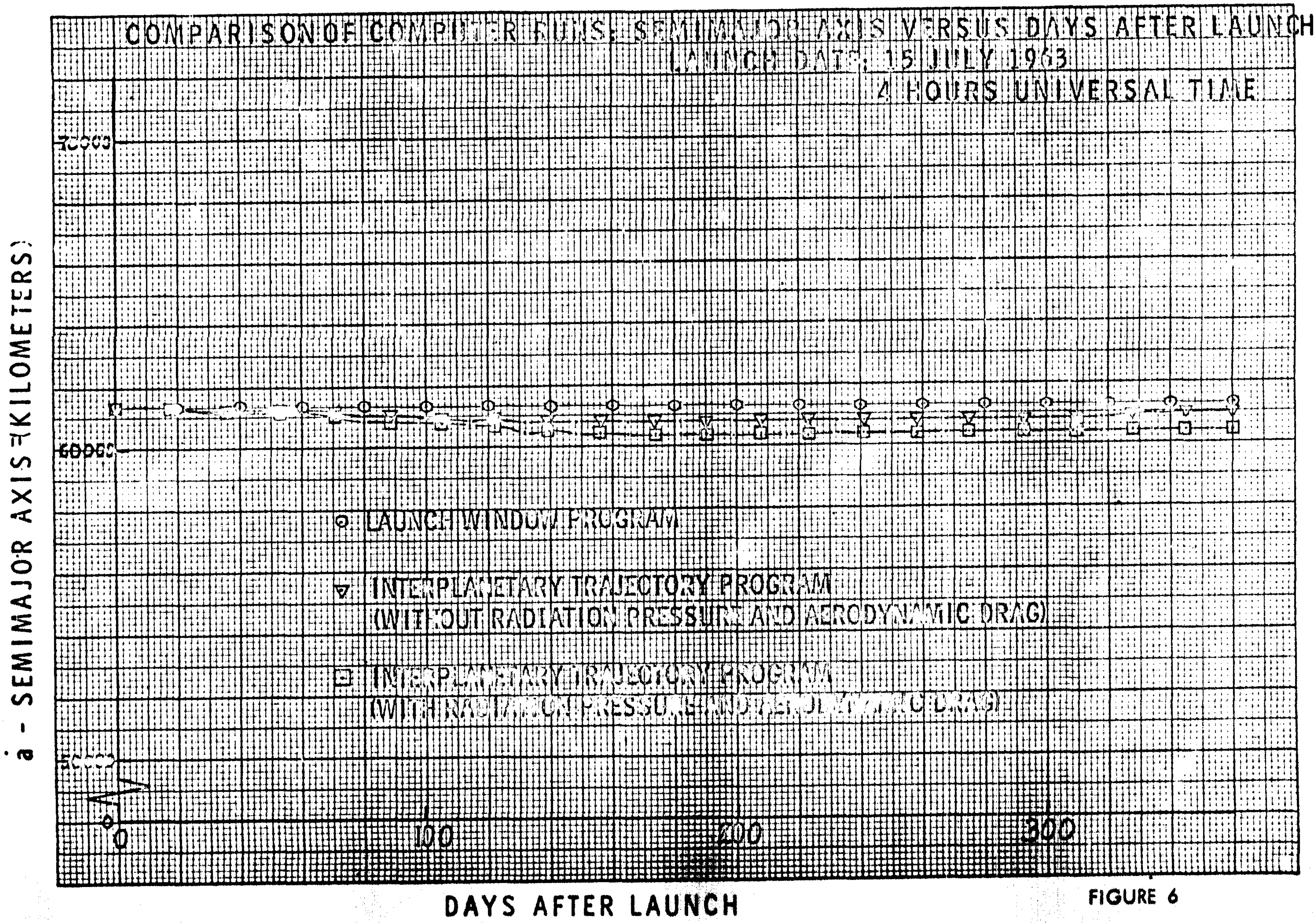
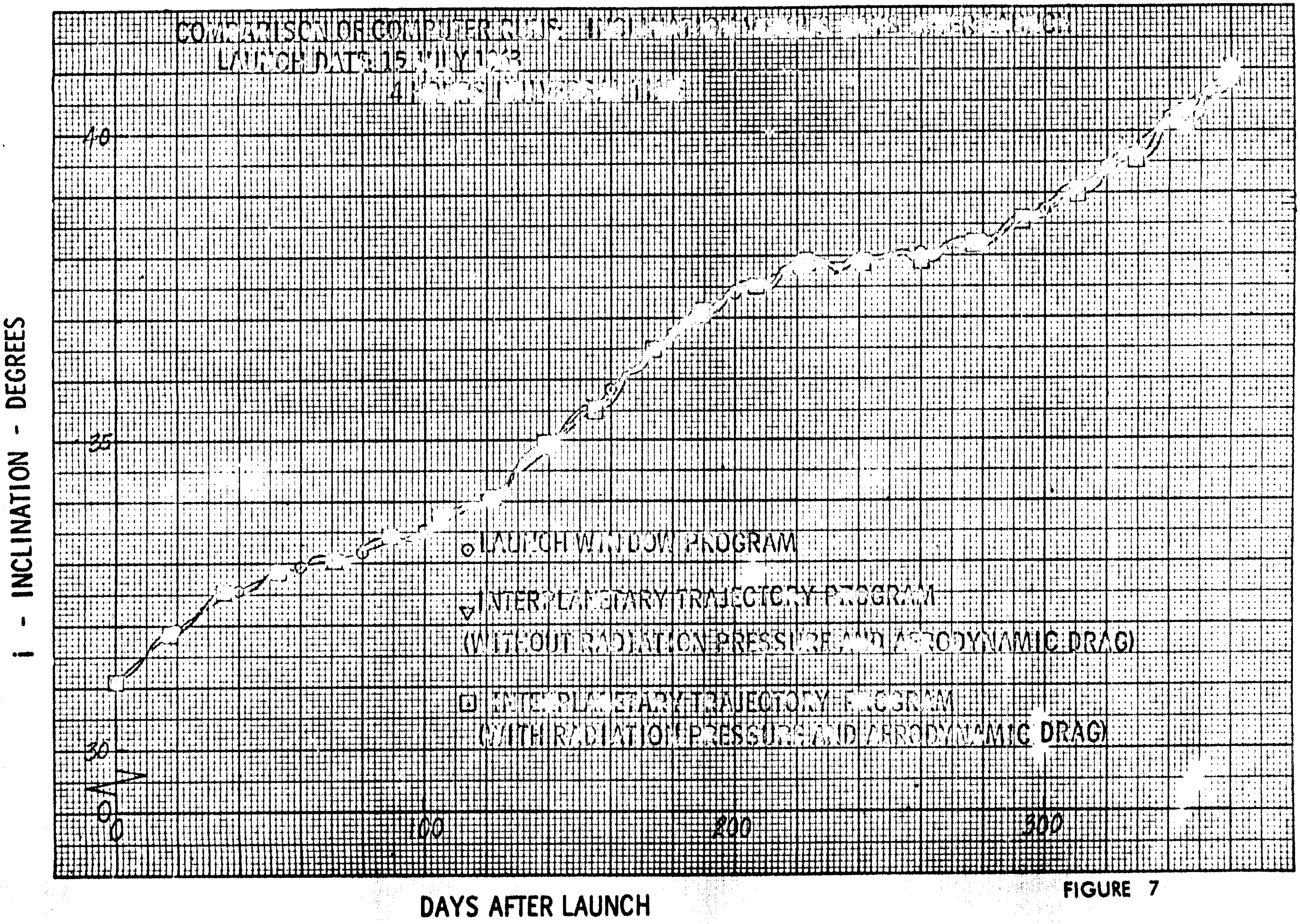

COMPARISON OF COMPUTER RUNS: SEMIMAJOR AXIS VERSUS DAYS AFTER LAUNCH
LAUNCH DATE: 15 JULY 1963
4 HOURS UNIVERSAL TIME
70000
60000
50000
LAUNCH WINDOW PROGRAM
INTERPLANETARY TRAJECTORY PROGRAM
(WITHOUT RADIATION PRESSURE AND AERODYNAMIC DRAG)
INTERPLANETARY TRAJECTORY PROGRAM
(WITH RADIATION PRESSURE AND AERODYNAMIC DRAG)
0
100
200
300
a - SEMIMAJOR AXIS (KILOMETERS)
DAYS AFTER LAUNCH

FIGURE 6



LAUNCH WINDOW PROGRAM
INTERPLANETARY TRAJECTORY PROGRAM
(WITHOUT RADIATION PRESSURE AND AERODYNAMIC DRAG)
INTERPLANETARY TRAJECTORY PROGRAM
(WITH RADIATION PRESSURE AND AERODYNAMIC DRAG)
40
35
30
0
0
100
200
300
i - INCLINATION - DEGREES
DAYS AFTER LAUNCH

FIGURE 7

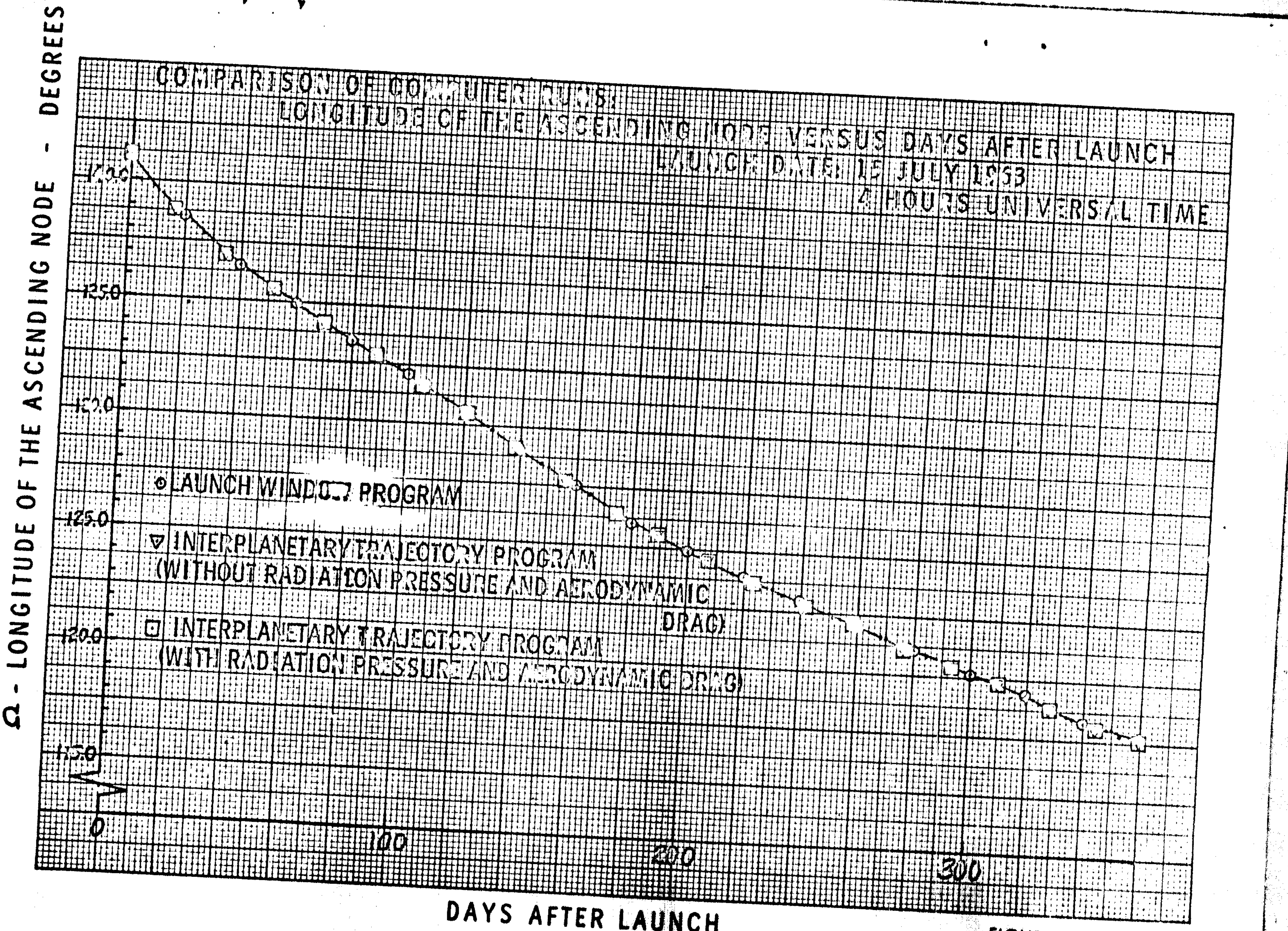

COMPARISON OF COMPUTER RUNS:
LONGITUDE OF THE ASCENDING NODE VERSUS DAYS AFTER LAUNCH
LAUNCH DATE: 15 JULY 1963
4 HOURS UNIVERSAL TIME
LAUNCH WINDOW PROGRAM
INTERPLANETARY TRAJECTORY PROGRAM
(WITHOUT RADIATION PRESSURE AND AERODYNAMIC DRAG)
INTERPLANETARY TRAJECTORY PROGRAM
(WITH RADIATION PRESSURE AND AERODYNAMIC DRAG)
Ω - LONGITUDE OF THE ASCENDING NODE - DEGREES
0
100
200
300
DAYS AFTER LAUNCH

FIGURE 8

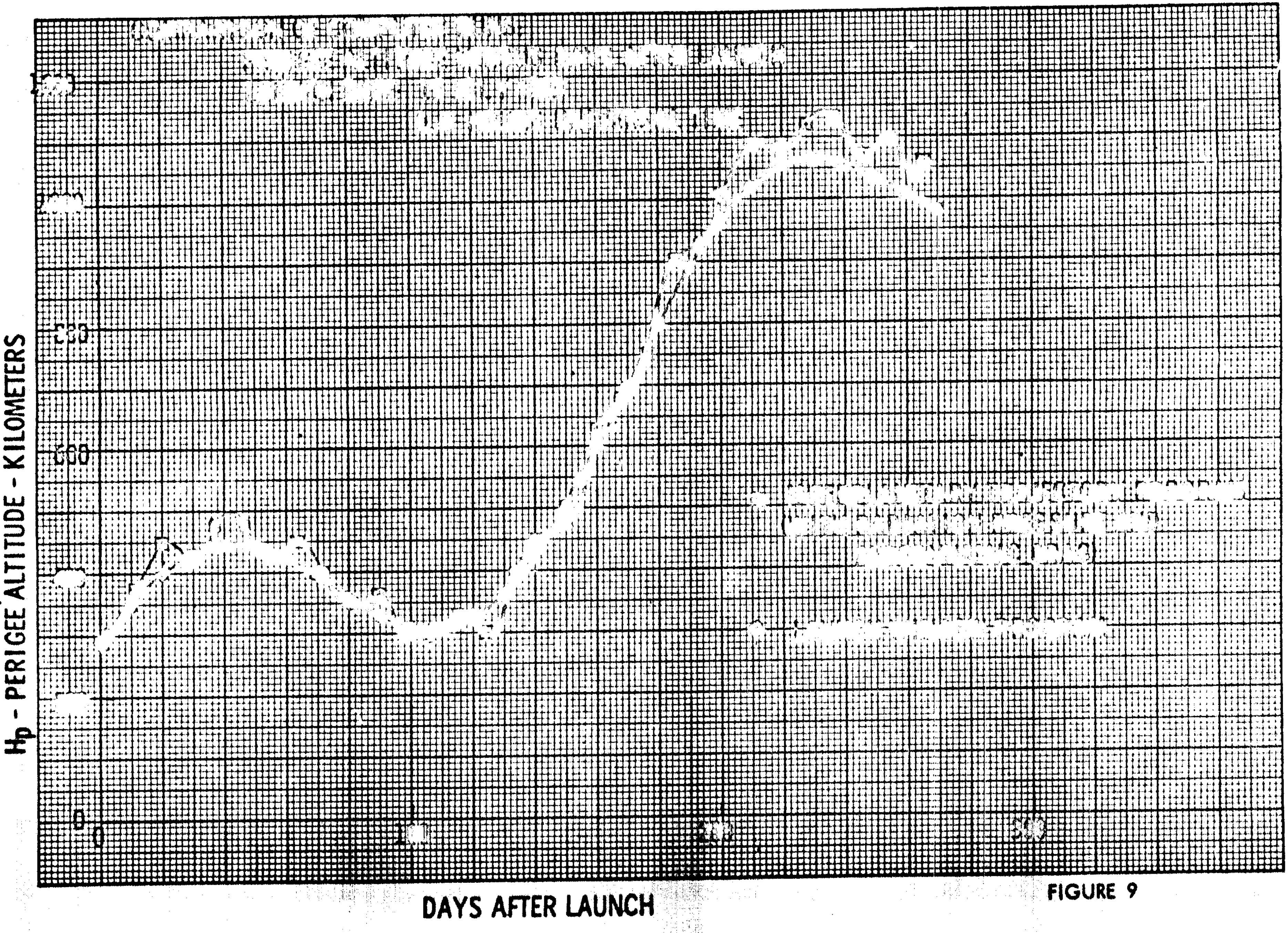
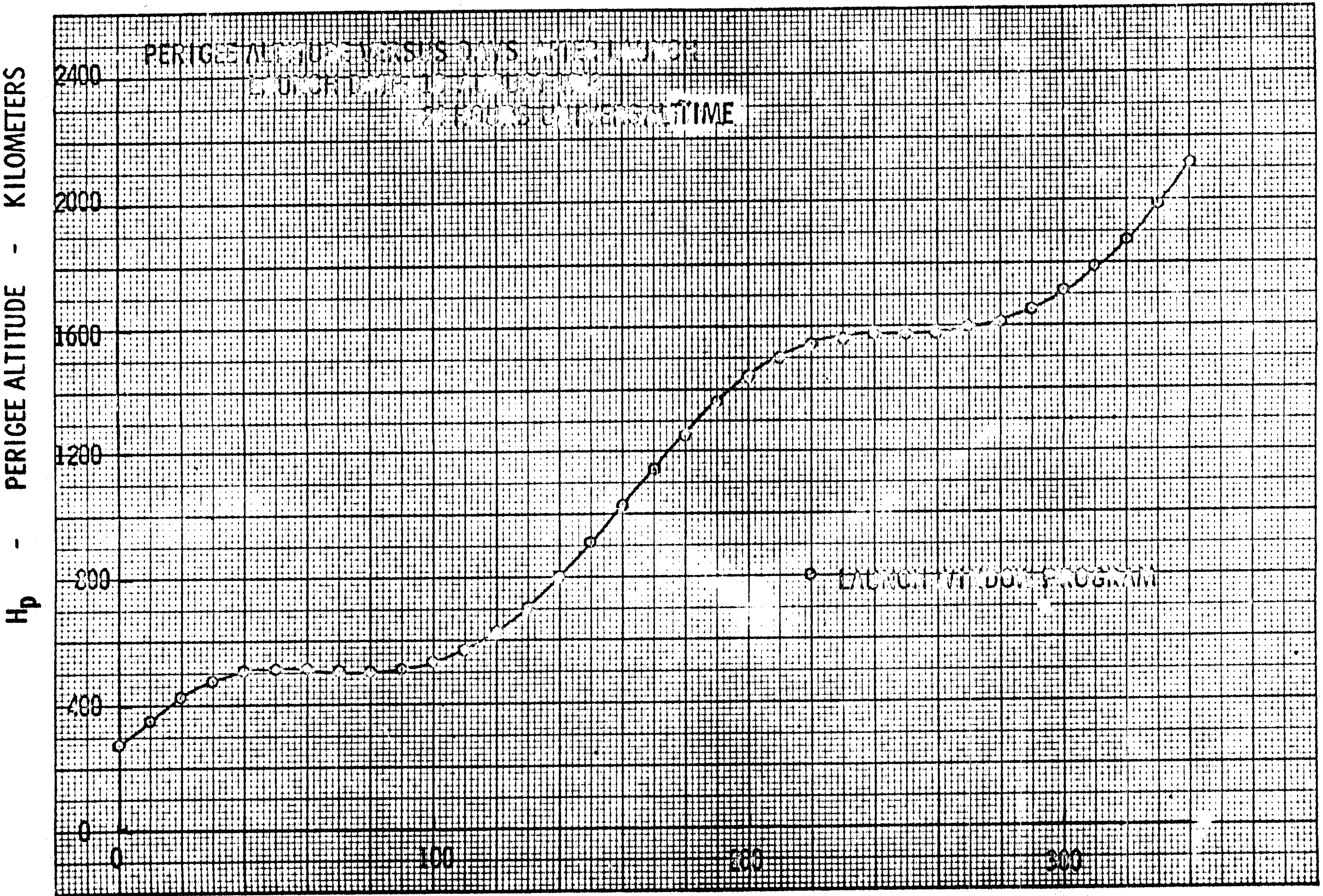
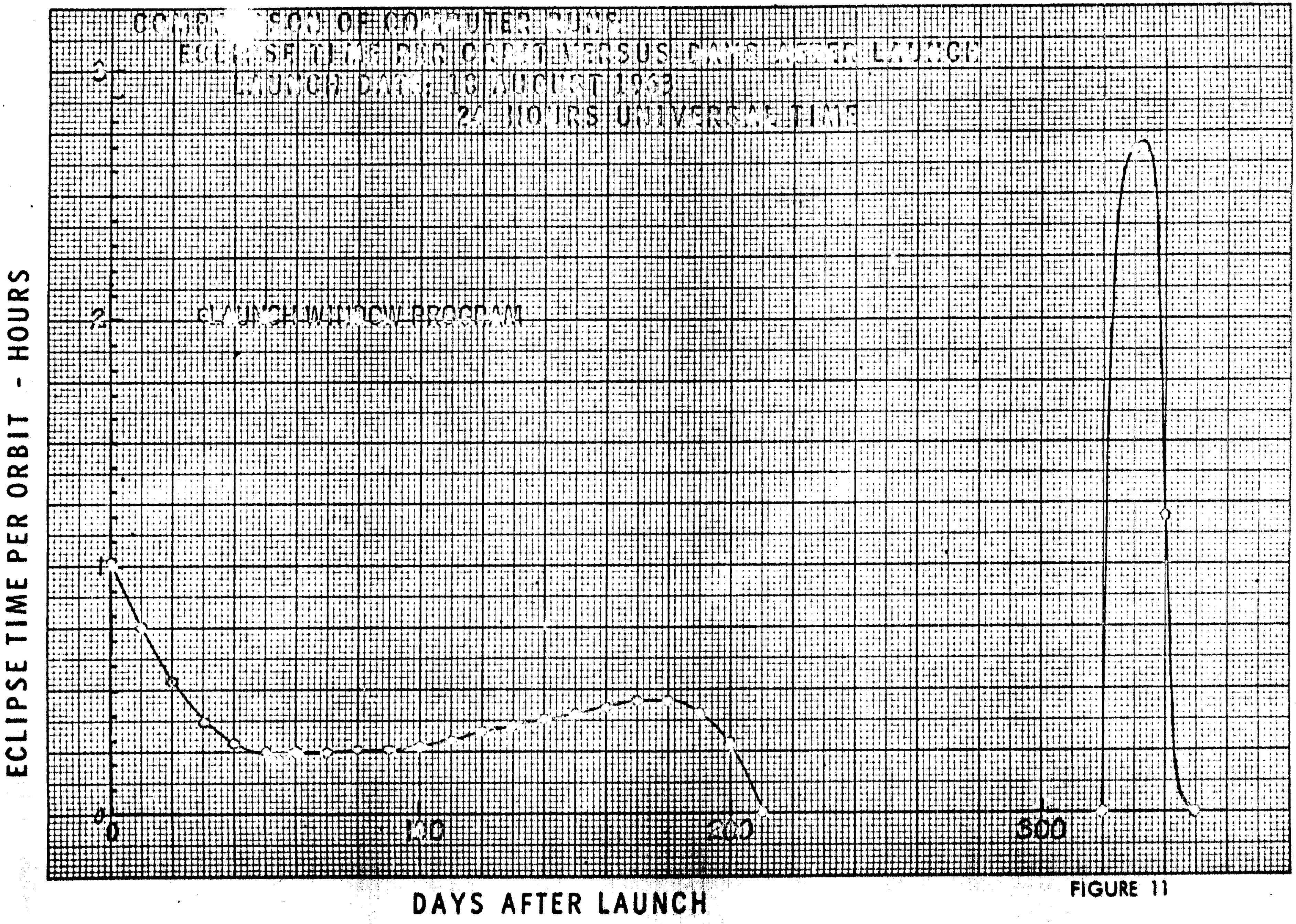

H_p - PERIGEE ALTITUDE - KILOMETERS
DAYS AFTER LAUNCH
0
100
200
300
600

FIGURE 9



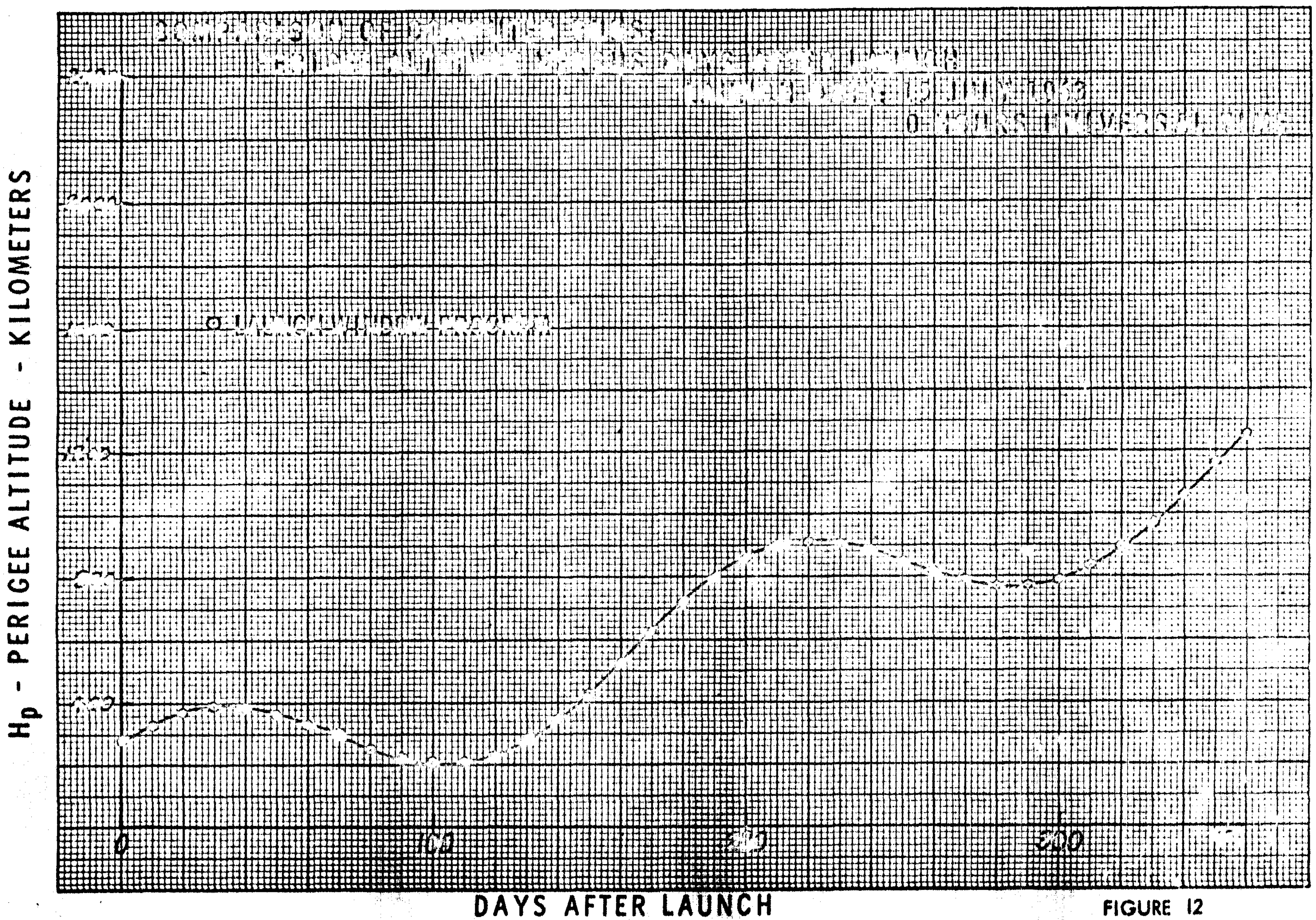
PERIGEE
TIME
2400
2000
1600
1200
800
400
0
0
100
200
300
H_p - PERIGEE ALTITUDE - KILOMETERS
DAYS AFTER LAUNCH

FIGURE 10



OF COMPUTER RUNS
VERSUS
LAUNCH DATE: 16 AUGUST 1963
24 HOURS UNIVERSAL TIME
LAUNCH WINDOW PROGRAM
ECLIPSE TIME PER ORBIT - HOURS
0
1
2
3
0
100
200
300
DAYS AFTER LAUNCH

FIGURE 11



Hp - PERIGEE ALTITUDE - KILOMETERS
0
100
200
300
DAYS AFTER LAUNCH

FIGURE 12

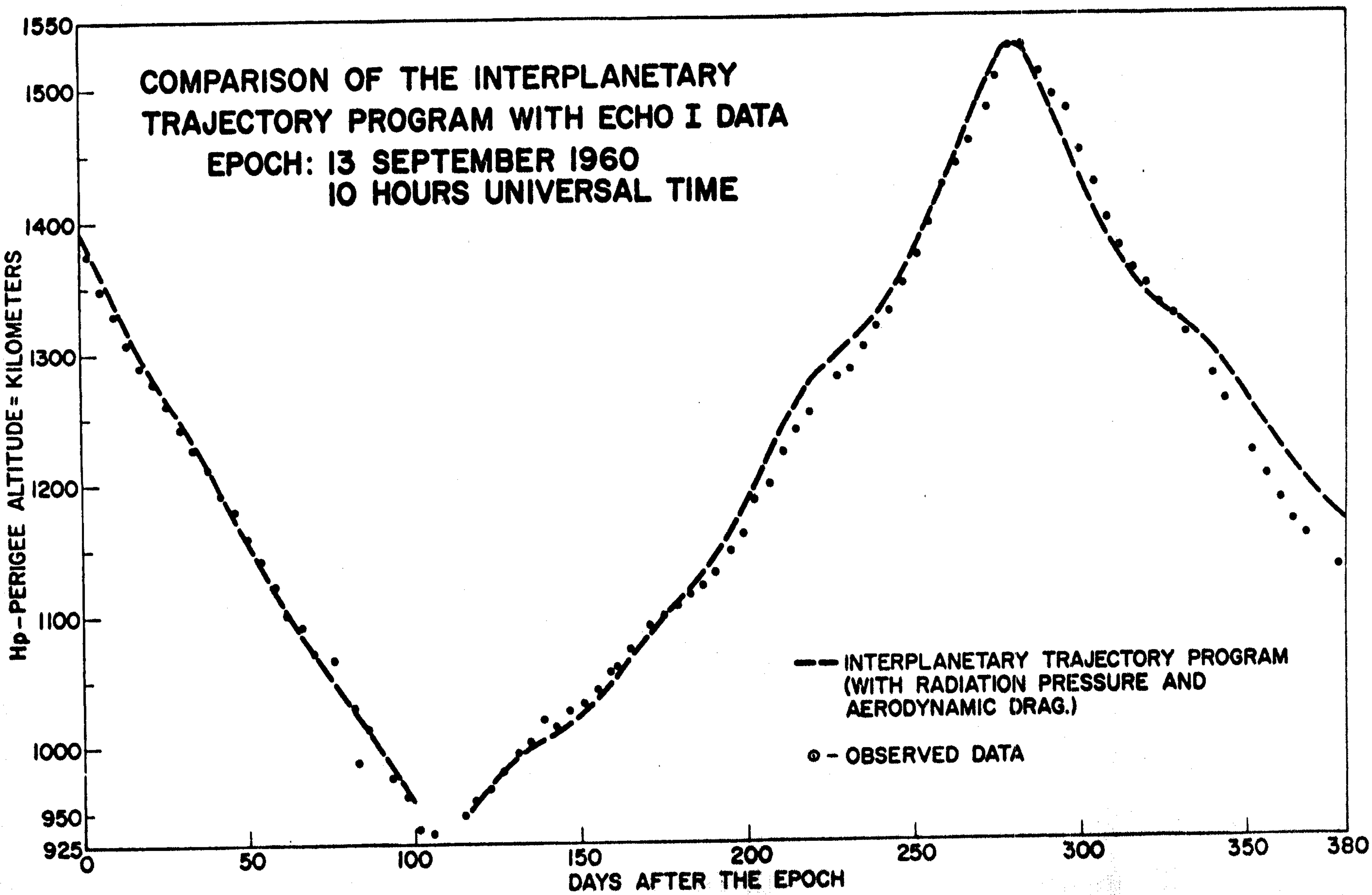

COMPARISON OF THE INTERPLANETARY
TRAJECTORY PROGRAM WITH ECHO I DATA
EPOCH: 13 SEPTEMBER 1960
10 HOURS UNIVERSAL TIME
Hp-PERIGEE ALTITUDE = KILOMETERS
1550
1500
1400
1300
1200
1100
1000
950
925
0
50
100
150
200
250
300
350
380
DAYS AFTER THE EPOCH
INTERPLANETARY TRAJECTORY PROGRAM
(WITH RADIATION PRESSURE AND
AERODYNAMIC DRAG.)
- OBSERVED DATA

FIGURE 13

TABLE I

EFFECTS OF INJECTION TOLERANCES ON LAUNCH WINDOW for July 15, 1963.

Injection Parameters	3 Variation	Lower Perigee Boundary (U.T. Hours)	Upper Perigee Boundary (U.T. hours)	Lower Shadow Boundary (U.T. Hours)	Upper Shadow Boundary (U.T. Hours)
Geocentric Altitude	+ 48017	0.234	9.38	2.270	4.50
$\Delta h \sim$ feet	- 48017	0.370	8.95	2.30	4.56
Azimuth	+ 0.323	0.317	9.18	2.259	4.535
$\Delta A_c \sim$ degrees	- 0.323	0.283	9.22	2.285	4.525
Geocentric Longitude	+ 0.492	0.268	9.17	2.245	4.50
$\sim$ degrees	- 0.492	0.332	9.23	2.31	4.56
Elevation Angle	+ 0.098	0.312	9.22	2.285	4.556
$E \sim$ degrees	- 0.098	0.281	9.18	2.27	4.51
Speed	+45.9	0.28	9.26	2.27	4.50
$V \sim$ feet/sec.	-45.9	0.31	9.14	2.30	4.56
Geocentric Latitude	+ 0.284	0.30	9.25	2.27	4.57
$\sim$ degrees	- 0.284	0.30	9.136	2.282	4.51
Nominal Case		0.30	9.2	2.28	4.53