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Radiation Exposure and Performance of Multiple Burn LEO-GEO Orbit Transfer Trajectories

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SUMMARY

Many potential strategies exist for the transfer of spacecraft from low earth orbit (LEO) to geosynchronous (GEO) orbit. However, only one has generally been utilized, that being a single impulsive burn at perigee and a GEO insertion burn at apogee. Multiple burn strategies have been discussed for orbit transfer vehicles (OTVs) but the transfer times and radiation exposure, particularly for potentially manned missions, have been used as arguments against those options. This paper presents quantitative results concerning the trip time and radiation encountered by multiple burn orbit transfer missions in order to establish the feasibility of manned missions, the vulnerability of electronics, and the shielding requirements. The performance of these multiple burn missions is quantified in terms of the payload and propellant variances from the minimum energy mission transfer. The missions analyzed varied from one to eight perigee burns and ranged from a high thrust, 1 g acceleration, cryogenic hydrogen-oxygen chemical propulsion system to a continuous burn, 0.001 g acceleration, hydrogen fueled resistojet propulsion system with a trip time of 60 days.

INTRODUCTION

The goal of mission analysis is to maximize payload capability by minimizing the fuel required. For orbit transfer vehicles this involves optimizing burn times as well as thrust direction. Theoretically, optimizing burn time includes the freedom to use multiple perigee burn trajectories in order to reduce the burn arc and approach the ideal impulsive burn, but the increased trip time and radiation exposure for potential manned missions have been used as arguments against their consideration. A well-publicized exception to the single burn scenario was the unmanned TDRSS communication satellite (ref. 1). In that instance, a failure occurred in the prime propulsion system and the spacecraft was put into an elliptical orbit having a perigee of 14 000 miles and an apogee of 22 500 miles. The spacecraft was then forced to rely on the low thrust auxiliary propulsion system to reach its final orbital destination. A multiple burn orbit transfer strategy was used (39 apogee burns)(ref. 2).

The present space mission models (refs. 3 to 6) incorporate missions that could benefit from or require a limit on thrust or acceleration. Large lightweight structures will be erected or deployed in low orbit and subsequently transferred to a higher or geosynchronous orbit. These structures are inherently acceleration limited and significant payload advantages could be achieved through the use of a multiple burn orbit transfer strategy.

Generally, the minimum fuel case for a circle-to-circle orbit transfer uses the "Hohmann transfer" involving two impulsive (instantaneous velocity change) burns, one at perigee and one at apogee. The impulses are applied

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parallel to the local horizontal with a component in the direction of any potential plane change. The ideal Hohmann transfer has infinite acceleration with zero burn time. In actuality, as the acceleration is reduced, the burn arcs become longer. Since thrust is applied away from the optimum points (perigee and apogee) the burns become less efficient. The total velocity increment required for the orbit transfer

$$\Delta V = \int_0^{t_b} A(t)dt,$$

where A is the acceleration and t_b is the burn time, increases above the impulsive case. The difference $\Delta V - \Delta_V$ impulsive is referred to as gravity loss, a measure of efficiency.

For the cases involving accelerations (thrust/mass ratio) less than 0.25 g, the standard two burn transfer begins to incur long burn arcs resulting in large gravity losses. But it is possible to reduce these losses and approach the impulsive case provided the penalties of longer trip time and increased radiation exposure can be tolerated. For any given thrust level, the total engine burn time is relatively fixed, therefore, by subdividing the burn time into several intermediate burns, each having a proportionately smaller arc length, the total gravity loss is reduced by approximately $1/N^2$, where N is the number of burns. Perigee burns contribute the greater portion of the gravity loss, therefore, subdividing the perigee burn produces the larger savings. It has been shown by Redding and Breakwell (ref. 7) that the optimal transfer incorporates more perigee burns than apogee burns. Each burn produces an intermediate orbit which increases trip time by approximately the period of that orbit. A computer program (ref. 8) was developed at Stanford University under NASA LeRC sponsorship to determine burn time, thrust direction history, total time and losses for the circle-to-circle transfer given the initial acceleration, Isp, number of burns, initial and final orbit states. That program was used to define the trajectories in this study effort.

Radiation exposure times for multiple perigee burn low thrust orbit transfer missions are inherently higher than those using single burn strategies. However, no correlations have been made between the radiation models and those trajectories to define the types of radiation and the dose rates which will be encountered. The objective of this study is to establish whether manned missions to GEO using multiple burn strategies are feasible and the shield thicknesses required. Vulnerability of electronics to radiation was also evaluated. In the case of manned missions, trip time which affects total dosage is the main issue. Electronics has the added concern of random particle interactions, but that was not addressed in this study.

Natural space radiation consists of galactic cosmic radiation, solar proton radiation and geomagnetically trapped (Van Allen) radiation. A discussion of these sources is provided in references 9 and 10. Since spacecraft involved with traversing the LEO-GEO corridor are affected by the radiation environment, it is in the best interest of the mission planners to utilize trajectories which minimize the radiation exposure to crewmen and on-board equipment. This paper is intended to quantify the radiation doses to be encountered by spacecraft using several orbit transfer strategies, but does not optimize the trajectory. Optimization to reduce radiation effects is heavily mission specific and must be performed on a case-by-case basis.

APPROACH

This study was conducted primarily to determine the radiation level which might be encountered by an orbit transfer vehicle performing a mission between low earth orbit and geosynchronous orbit. It was necessary to identify the differences between the conventional single perigee burn trajectory and multiple perigee burn trajectories since those differences can significantly impact the design of future OTVs. They affect the low acceleration capability of the vehicle, the engine thrust level, and safety in the case of manned missions. If multiple burns are accepted as a viable option, that strategy can be utilized as a fail operational/fail safe backup position for the current mode of operation.

Eight specific cases were run, outlined in table I, to establish mission radiation effects. Cases 1 to 3 were the base cases using the single perigee trajectory to determine the sensitivities to payload mass and acceleration. Cases 4 to 6 were used to define the effect of multiple perigee burns, while case 8 was considered as the upper limit, it being a continuous low thrust 60 day burn making approximately 405 orbits of the Earth during the transit. Case 7 was an attempt to evaluate the sensitivity of transit through the inner Van Allen Belt by performing an intermediate apogee circularization burn at an altitude between the two major Van Allen Belts, see figure 1, at approximately 10K km, and then performing another perigee burn to complete the vehicle transfer to GEO. Cases 1 to 7 used a high performance cryogenic hydrogen-oxygen chemical propulsion system having a specific impulse (Isp) of 485 sec, while case 8 used a hydrogen fueled resistojet with an Isp of 830 sec. These cases bound the expected range of candidate mission strategies and the results can be utilized to establish feasibility of this concept.

No attempt was made in this study to optimize the trajectories in order to reduce the radiation effects. Since the proton and electron flux around the Earth varies with longitude and latitude as well as altitude, minimizing radiation effects requires a tradeoff between propellant usage, trip time and engine burn time. If the radiation limitations for a specific mission are extremely critical, optimum trajectories for these radiation constrained missions can be obtained through controls on the location and length of the engine burns.

In addition to the analysis of the eight specific missions, a parametric analysis was performed using the Stanford computer program. Trajectories were defined for 1, 2, 4, and 8 perigee burn LEO-GEO orbit transfer missions over the acceleration range 0.02 to 1.0 g. That program has the capability of operating the propulsion system in either of two modes - fixed thrust or fixed acceleration. Fixed thrust is the conventional operating mode for current systems. Maximum acceleration is reached at the end of the final burn, when mass is least. Therefore, if an acceleration limit is imposed, the fixed thrust mode would be less efficient than the fixed acceleration mode. Only the data for cases of fixed acceleration are presented here. Thrust mode had only a secondary effect on trip time and radiation exposure.

Optimal fuel efficient trajectories were defined in order to establish the mission ΔV requirements. These requirements were then used to determine engine burn time, trip time, propellant requirements, and the resultant payload capability. Results are expressed in terms of increased propellant and decreased payload due to gravity losses as a function of the acceleration and number of perigee burns.

A temporal and spatial history was obtained for each of the eight cases analyzed. These data were then correlated using the latest proton and electron radiation environment models (refs. 11 and 12) by a computer technique developed and operated by the Johnson Space Center (JSC), to determine dose rates as a function of radiation type as well as total dose. Reference 13 contains a description of the Orbital Radiation Program (ORP) used by JSC. For this study, it was assumed that solar flares did not occur during an orbit transfer. Proton, electron, and "Bremsstrahlung" dose radiation were calculated as a function of shielding thickness for each mission. Shield thickness was treated as a parameter and varied from an areal density of zero to 10 g/cm^2, the equivalent of a 3.7 cm thick sheet of aluminum.

In an actual mission, a significant amount of time would be spent in geosynchronous orbit. Therefore, the JSC program was also used to calculate the expected daily radiation dose at the GEO location as a function of shield thickness. Total mission dose is the sum of the transfer and on-station radiation. The resultant radiation exposures can be used to establish requirements for the protection of astronauts and electronics.

RESULTS AND DISCUSSION

LEO-GEO Orbit Transfer Trajectories

Figure 2 presents two strategies for LEO to GEO orbit transfer trajectories. Figure 2(a) describes the classical high thrust Hohmann transfer while figure 2(b) depicts a low thrust, multiple perigee burn planar LEO to GEO orbit transfer trajectory. In the latter case there are four burns, one after each revolution of the spacecraft around the Earth. Each perigee burn (ΔV_{pi}) raises the apogee altitude until geosynchronous altitude is reached after the fourth burn. Thrust is then applied at apogee (ΔV_a) to circularize the orbit. Gravity losses (g-loss) above the optimum impulsive (Hohmann) transfer are introduced by the noninfinitessimal burn arc. However, through the use of multiple perigee burns the length of the burn arc is reduced, thereby keeping g-losses to a minimum. For a given thrust level the total burn time is essentially constant, therefore, the length of the burn arc is inversely proportional to the number of burns. In an optimized trajectory the length of each burn would vary.

Effect of Acceleration Level on Burn Time and Trip Time

Figure 3 shows the variation of trip time and burn time as a function of acceleration for one, two, four, and eight perigee burn trajectories. Trip time varied from approximately 5 hr for the single burn case to 28 hr for an eight perigee mission while burn time reached as much as 4 hr for the very low (0.03 g) acceleration missions. Mission trip time is seen to be almost independent of the acceleration level for a given orbit transfer strategy (fixed number of burns). The variance between cases one, two, and three is within ±2 percent, therefore, the payload for cases four through eight was fixed to simplify the analysis. Since acceleration is simply the ratio of engine thrust level to initial system mass (payload + vehicle + propellant), it can be concluded that for a given trajectory strategy, thrust level has only a second order effect on trip time. However, for a given vehicle and payload, burn time is an inverse function of acceleration, or thrust level. And that function is independent of the trajectory strategy. The significance of this fact is that a given engine can be operated over a range of trajectories without significantly affecting its operating life, although the effect produced by the start-stop cycles must be considered.

Multiple Perigee Burn LEO-GEO Performance Benefits

Figure 4 displays the potential mission performance benefits associated with multiple perigee burns. The curves emphasize the effect of increasing the number of perigee burns on gravity losses. For example, in the case of a mission involving a fixed acceleration of 1/16 g; the single perigee case is prohibitive in payload and propellant costs; the two burn case uses 2.7 percent more propellant resulting in 5.3 percent less payload; while the eight burn case uses only 0.2 percent more propellant and 0.4 percent less payload than the ideal Hohmann transfer. The ratio of payload decrease to propellant increases will vary as an inverse function of the engine Isp, since Isp is the primary factor in the establishment of vehicle propellant requirements, e.g., an Isp = 450 sec produces a ratio of approximately 3:1, compared to the 2:1 ratio shown for the Isp = 485 sec.

This analysis does not include any propellant losses due to boiloff, prefire engine conditioning, or shutdown residuals. These losses are very dependent on engine and vehicle design and operation. As a point of reference, a study of Large Space Structures (ref. 14) calculated these losses to be approximately 5 percent for a nine burn mission.

Radiation Dose Comparison

Radiation dose for the eight missions defined in table I was calculated using the orbit transfer trajectories and the latest AP-8 proton (ref. 11) and AEI-7 electron (ref. 12) models. Dosage was calculated for a range of shielding thicknesses from 0 to 10 g/cm^2. This unit, when divided by material density, defines the thickness requirement for that material, e.g., 1.0 g/cm^2 = 0.37 cm of aluminum equivalent. The dosages were calculated in terms of the proton, electron and Bremsstrahlung contribution. Table II presents the summation of these contributions for the eight cases as well as the daily radiation dose at GEO. As points of reference for the shield thickness column of table II; the Apollo astronaut space suit corresponded to roughly 0.15 g/cm^2, the Shuttle is approximately 1.5 g/cm^2, while the Apollo spacecraft had the equivalence of a 9.0 g/cm^2 thick shield. As expected the radiation dose decreases dramatically with increased shielding and increases almost exponentially with trip time.

NASA presently uses the radiation exposure guidelines in table III established by the National Academy of Sciences (ref. 15) for men in space. Tables II and III can be related using the conversion that radiation dose in rem (Roentgen-equivalent man) = radiation in rads multiplied by the RBE (Relative Biological Effectiveness). The values of RBE for various types of radiation is presented in reference 16. An average value for RBE of 1.2 has been used in this study based on other system studies encountering similar types of radiation. For that value of RBE it can then be seen that with a moderate spacecraft shield thickness, 1.5 to 2.0 g/cm^2, a manned sortie mission using up to four perigee burns could be safely completed. Special provisions for astronaut safety are necessary if a very low acceleration mission using a higher number of burns were required.

An interesting phenomena is displayed in the last column of table II, GEO dose/day. For a given shield thickness the daily dose at GEO is roughly equivalent to the radiation experienced by a one perigee burn mission making a single pass through the Van Allen Belts. And a week at GEO is equivalent to the radiation dose for a two perigee burn orbit transfer mission with a moderate amount of shielding. The resultant conclusion is that for "man in GEO" the radiation constraint will most likely be "stay time on orbit" rather than the orbit transfer.

Case 7, which used a two perigee burn strategy with only a single passage through the inner Van Allen Belt, did not provide any radiation benefits. Although analysis of the trajectory did indicate the potential for optimization to minimize radiation effects, the gains may not be substantial and would cause complexities in an actual mission.

Case 8, the very low thrust, low acceleration, resistojet propulsion mission had a trip time of 60 days. As a primary propulsion system for manned missions the radiation dosage was unacceptable at any level of shielding. However, for unmanned missions requiring this type propulsion, e.g., large systems with acceleration constraints or high mass systems with vehicle/propellant quantity constraints, the radiation encountered using only a shield thickness of 1.0 g/cm^2 is well within the tolerance of 10^5 rad for state-of-the-art space qualified electronics.

Figure 5 details some of the data for case five and is presented as representative of the constituents of the total radiation dose. The predominant radiation contribution is from the heavier protons. With no shielding the only other component is the electrons, but as shielding is added, a portion of the electrons are stopped and form secondary particles, Bremsstrahlung radiation. However, at approximately 2 g/cm^2 the electrons are completely stopped and make no further contribution. The Bremsstrahlung contribution goes through a maximum and then decreases as shield thickness increases. Total radiation data shown in table II is simply the summation of these components.

CONCLUSIONS

A trajectory model which produced a time-location history for LEO to GEO orbit transfer missions in combination with a radiation environment model was used to predict the radiation exposure encountered by the spacecraft. The results indicate that with a moderate amount of shielding, approximately 1.5 to 2.0 g/cm^2, comparable to that of the Shuttle, astronauts can safely perform LEO-GEO orbit transfer missions using a multiple perigee burn strategy. Radiation dose/day at GEO was found to be comparable to the total dose for the single perigee burn mission. While a week on orbit at GEO was equivalent to the two perigee burn case. The radiation constraints for manned GEO missions will, therefore, be significantly affected by the stay time in orbit rather than merely the trajectory strategy. There does not appear to be any constraints imposed on unmanned missions due to the radiation sensitivity of the electronics, even for the case of very low acceleration missions. These factors open up an added dimension for the mission planners in terms of enhancements to mission flexibility and system reliability.

Performance benefits of multi-burn orbit transfer missions were determined. Multiple burn missions can be used with small (1 percent) payload penalties in comparison with more conventional, high thrust impulsive orbit transfers. And for missions requiring accelerations less than 0.2 g, it has a clear advantage in payload capability. In fact, for some acceleration limited missions such as very large structures, multiple burns may be an enabling concept. Total trip time is only increased from approximately 5 hr for a single burn mission to 28 hr for the mission using as many as eight perigee burns. Therefore, trip time will probably not be a discriminating factor in the selection of mission strategy.

An additional benefit or degree of freedom for missions incorporating multi-burn capability is the probability of mission success with an engine out. Should an engine fail, the mission could be completed with the original launch propellant load. An on-board control system would be required to track the propellant usage to each engine and to optimize the trajectory. But the payload/mission could be performed at minimal penalty.

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TABLE I. TRAJECTORY CASES

Case number	Strategy	Payload mass, lb	Acceleration, g	Trip time, hr
1	1 perigee 1 apogee	32 400	1	5.3
2	1 perigee 1 apogee	7 340	1/4	5.5
3	1 perigee 1 apogee	32 400	1/2	5.4
4	2 perigee 1 apogee	7 340	1/4	8.2
5	4 perigee 1 apogee	7 340	1/4	14.7
6	8 perigee 1 apogee	7 340	1/8	28.7
7	2 perigee 2 apogee	7 340	1/4	12.8
8	Continuous	7 340	<0.001	1400

TABLE II. RADIATION DOSE[a] COMPARISON

Shield[c] thickness, g/cm2	Trajectory case								GEO/[b] day
	1	2	3	4	5	6	7	8	
0.0	1.44 E07	1.84 E07	1.57 E07	1.64 E07	7.13 E07	1.32 E08	1.34 E08	7.99 E09	6.04 E04
0.1	1.28 E03	1.48 E03	1.36 E03	5.22 E03	6.29 E03	1.35 E05	1.01 E04	2.00 E06	4.85 E03
0.5	7.70 E01	9.61 E01	8.31 E01	2.28 E02	3.50 E02	8.49 E02	4.88 E02	1.19 E05	1.35 E02
1.0	8.13 E00	1.05 E01	8.80 E00	2.42 E01	5.74 E01	1.33 E02	7.11 E01	2.11 E04	5.95 E00
2.0	4.04 E-01	3.96 E-01	3.87 E-01	3.42 E00	1.31 E01	2.42 E01	1.10 E01	6.27 E03	3.15 E-01
5.0	1.31 E-01	9.49 E-02	1.11 E-01	7.89 E-01	6.18 E00	1.01 E01	4.37 E00	2.91 E03	1.45 E-01
10.0	7.28 E-02	5.19 E-02	6.15 E-02	2.90 E-01	3.79 E00	5.79 E00	2.58 E00	1.75 E03	7.93 E-02

[a]Dose in rads (proton + electron + Bremmstrahlung).
[b]Dose in rads/day.
[c]1.0 gm/cm^2 = 0 37 cm aluminum equivalent.

TABLE III. NASA RADIATION EXPOSURE GUIDELINES (REM)

[Limits established by National Academy of Sciences 1970 for Man in Space.]

Time	Blood forming organs	Skin	Eye	Testes
Penetration depth	5 cm	0.1 mm	3 mm	3 mm
1 year av. daily	0.2	0.5	0.3	0.1
30 day maximum	25.0	75	37	13
90 day maximum	35.0	105	52	18
Yearly maximum	75.0	225	112	38
Career limit	400	1200	600	200

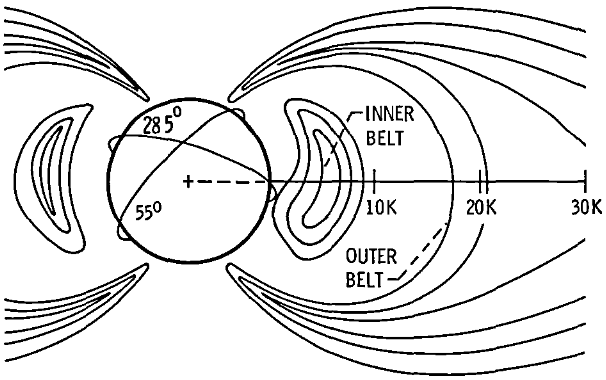
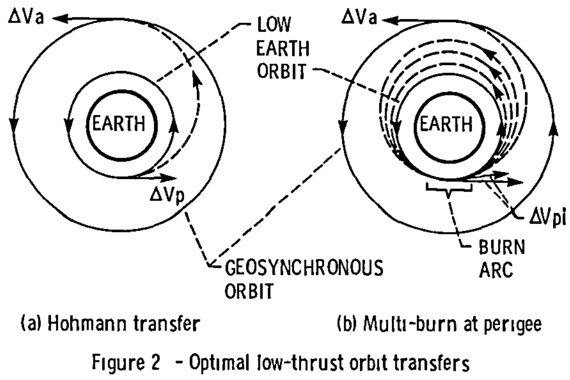


Figure 1. - Schematic of Van Allen radiation belts.



(a) Hohmann transfer

(b) Multi-burn at perigee

Figure 2 - Optimal low-thrust orbit transfers

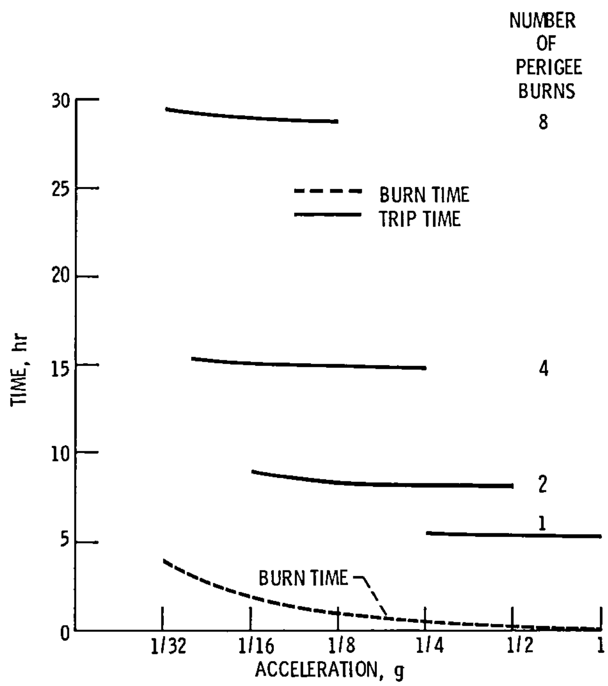


Figure 3 - Effect of acceleration on burn time and trip time for LEO TO GEO trajectories

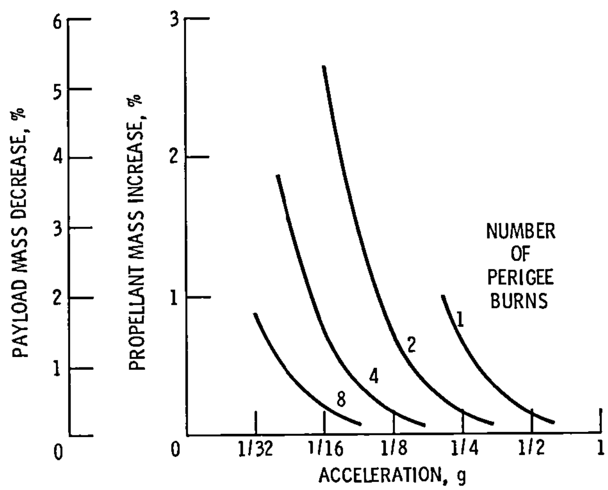


Figure 4. - Multiple perigee burn LEO-GEO performance variance from minimum energy transfer mission

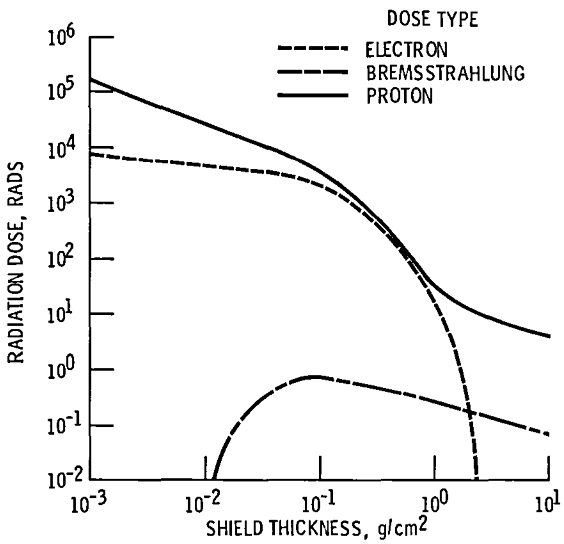


Figure 5 - Radiation dose constituents for a 4 perigee burn LEO-GEO orbit transfer mission

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16 Abstract

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