EGERTON UNIVERSITY

EEEN 567 – SATELLITE ENGINEERING

CONTINUOUS ASSESSMENT TEST 1

PART 1

Answer Two of the Following Five Questions

1. You are designing the communication subsystem for an Earth observation satellite. The mission requires the satellite to pass over the same point on Earth every 3 days to monitor changes. If the satellite is to be placed in a circular orbit, what altitude must it have? (Assume Earth's average radius = 6371 km, Earth's mass M = 5.972×10^{24} kg, and gravitational constant G = 6.674×10^{-11} $m^3/kg/s^2$).

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This requires matching the orbital period to a sub-cycle of Earth's rotation. A 3-day repeat cycle means the satellite completes an integer number of orbits in 3 sidereal days.

1. Sidereal day = 86164 seconds. 3 sidereal days = 258,492 seconds.
2. For a practical repeat track, choose a number of orbits (N) that fits this. For simplicity, let's find the period for exactly 1 orbit per 3 days: **T = 258,492 s**. However, typical repeat cycles use multiple orbits. Let's try **N=45 orbits in 3 days**: T = 258,492 / 45 ≈ 5744.3 seconds.
3. Use Kepler's Third Law for circular orbits: $T = 2\pi \sqrt{(a^3/\mu)}$, where $\mu = GM = 3.986 \times 10^{14}$ m^3/s^2.
4. Solve for semi-major axis *a*: a = [(T√μ) / (2π)]^(2/3)

   a = [(5744.3 * √(3.986e14)) / (2π)]^(2/3) ≈ 7,678 km.
5. Altitude = a - Earth's radius = 7678 km - 6371 km = ~1307 km.

2. Your satellite is in a circular Low Earth Orbit (LEO) at an altitude of 1000 km. For the worst-case (equatorial) orbit, what is the maximum duration of eclipse (time in Earth's shadow) the satellite will experience? How does this directly impact the design of the electrical power subsystem? (Ignore atmospheric refraction).

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Orbital Parameters: Semi-major axis a = 6371 + 1000 = 7371 km. Orbital Period $T = 2\pi \sqrt{(a^3/\mu)} \approx$ 6301 seconds (~105 minutes).

Geometry of Shadow: The eclipse occurs through the cylindrical shadow of Earth. The angular half-width of the shadow (α) seen from the satellite is: $\alpha = \arcsin(R\_Earth / a) = \arcsin(6371/7371) \approx 60.0°$.

Eclipse Fraction: For a circular orbit, the fraction of the orbit in eclipse is (α / 180°). α = 60°, so fraction = 60/180 = 1/3.

Maximum Eclipse Duration: (1/3) * Orbital Period ≈ (1/3) * 6301 s ≈ 2100 seconds, or 35 minutes.

Impact on EPS Design: The batteries must be sized to provide full power for at least 35 minutes. The solar array must be sized not only for average power but also to fully recharge the batteries during the ~70 minutes of sunlight, accounting for depth of discharge and charge efficiency. This directly determines the mass, volume, and cost of the solar panels and batteries.

3. A LEO (Low Earth Orbit) satellite operates in a circular orbit at an altitude of 800 km. Assuming a spherical Earth with a radius of 6371 km and standard gravitational parameter $\mu = 3.986\times10^5$ km^3/s^2, calculate:

   a) The orbital period in minutes.

   b) The free-space path loss (FSPL) in dB for a downlink at 8 GHz when the satellite is directly overhead (at zenith).

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a) **Orbital Period:**

The semi-major axis a = Earth radius + altitude = 6371 + 800 = 7171 km.

Orbital period $T = 2\pi \sqrt{(a^3/\mu)}$

$T = 2\pi \sqrt{((7171)^3 / (3.986\times10^5))} \approx 2\pi \sqrt{(3.686\times10^8 / 3.986\times10^5)} \approx 2\pi \sqrt{(924.7)} \approx 2\pi * 30.41$

T ≈ 191.0 seconds ≈ **1.02 hours ≈ 61.2 minutes**.

b) **Free-Space Path Loss:**

Distance d = altitude = 800 km = 8×10^5 m.

Frequency f = 8 GHz = 8×10^9 Hz.

Speed of light *c* = 3×10^8 m/s.

Wavelength λ = c/f = 0.0375 m.

FSPL (in dB) = $20 \log_{10}(4\pi d / \lambda)$

$= 20 \log_{10}((4\pi \times 8\times10^5) / 0.0375)$

$= 20 \log_{10}(2.68\times10^8) \approx 20 \times 8.428 \approx$ **168.56 dB**.

4. A small Earth-observation satellite uses a deployable solar array. In sunlight, the array provides 240 W at 28 V bus voltage. The satellite's average load is 150 W. During eclipse, a Lithium-ion battery supplies the load.

a) What is the required battery capacity in Watt-hours (Wh) if the maximum eclipse duration is 35 minutes?

b) If the battery depth of discharge (DoD) is limited to 30% for longevity, what is the *total* battery capacity needed in Wh?

c) From (a), what average current does the solar array provide to the bus?

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a) Energy needed during eclipse = Power × Time = 150 W × (35/60) h ≈ **87.5 Wh**.

b) Total battery capacity = (Energy needed) / DoD = 87.5 Wh / 0.30 ≈ **291.7 Wh**.

c) Average solar array current during sunlit period (for power delivery to bus & battery):

Array Power = Bus Power + Battery Charging Power. To find average current to the bus, we consider the bus load is supplied directly. The bus current from the array when supporting the load is: $I = P_{load} / V_{bus}$ = 150 W / 28 V ≈ **5.36 A**. (Note: Additional array current would simultaneously charge the battery).

5. A satellite electronic unit has a failure rate (λ) of 2×10^{-6} failures/hour. Assume a constant failure rate (exponential distribution).

a) What is the probability that this unit will survive a 5-year mission (43,800 hours) without failure?

b) If the satellite has 20 such identical units in series (where any single failure causes system failure), what is the system reliability over 5 years?

c) Briefly explain one passive and one active thermal control method used in satellites.

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a) Reliability $R(t) = e^{-\lambda t} = \exp(-2 \times 10^{-6} \times 43800) = \exp(-0.0876) \approx 0.916$ or 91.6%.

b) System Reliability $R_{sys}(t) = [R_{unit}(t)]^{20} = (0.916)^{20} \approx 0.167$ or 16.7%.

c) Passive: Multi-Layer Insulation (MLI) blankets. They consist of multiple reflective layers to drastically reduce radiative heat transfer, isolating the satellite from external temperature extremes.

Active: Electrical Heaters. They are controlled by thermostats or onboard computers to provide precise heating to components (like batteries and thrusters) that must stay within their operational temperature range.

6. The Attitude Determination and Control Subsystem (ADCS) is crucial for satellite pointing.

a) Name two common sensors used for attitude determination and briefly state what they measure.

b) Name two common actuators used for attitude control and state one key advantage of each.

c) Why is a reaction wheel said to be a "momentum exchange device," and what is the consequent problem that must be managed?

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(a) 1) Sun Sensors: Measure the vector direction to the sun relative to the spacecraft body. 2) Magnetometers: Measure the vector direction and strength of the Earth's magnetic field.

b) 1) Reaction Wheels/Momentum Wheels: Electrically driven rotors that change the satellite's attitude by accelerating or decelerating. *Advantage:* Provide precise, continuous control without expending propellant. 2) Magnetic Torquers (Torquerods): Current-carrying coils that create a magnetic dipole to interact with Earth's field. *Advantage:* No moving parts, extremely reliable, and use only electrical power (no propellant).

c) A reaction wheel changes the satellite's attitude by transferring angular momentum *between* the wheel and the satellite body, keeping the *total* angular momentum of the system constant. The consequent problem is momentum accumulation: External disturbance torques (e.g., from solar radiation pressure, gravity gradient) constantly add small amounts of momentum to the system. The wheels must spin faster to absorb this, eventually hitting their maximum speed (saturation). To desaturate them, another actuator like a magnetic torquer or thruster must be used to "dump" the excess momentum to the external environment.

PART 2

Answer All Questions

1. What primary physical quantity does a DME system measure to determine an aircraft's position?

- ◯ Azimuth (Bearing) relative to the ground station.
- ◯ Altitude above the ground station.
- ◯ Slant range distance between the aircraft and the ground station.
- ◯ Horizontal (ground) distance between the aircraft and the ground station.

2. The localizer transmitter of a standard ILS operates in which frequency band?

- ◯ 108.10 MHz – 111.95 MHz

◯ 329.15 MHz – 335.00 MHz

◯ 75 MHz

◯ 328.6 MHz – 335.4 MHz

The localizer operates in the VHF band from 108.10 to 111.95 MHz (on odd-tenth MHz frequencies only). Option D (328.6 MHz – 335.4 MHz) is the glide slope UHF band.

17. The localizer AM modulation frequencies (90 Hz and 150 Hz) produce a combined pattern where the on-course signal has ....

◯ Equal modulation depth of both tones

◯ Only 90 Hz modulation

◯ Only 150 Hz modulation

◯ 20% 90 Hz and 80% 150 Hz modulation

3. The primary function of a tracking radar, as opposed to a search radar, is to:

◯ Detect new targets over a large volume.

◯ Continuously measure the range, azimuth, and elevation of a specific target.

◯ Identify the type of target (e.g., aircraft, ship).

◯ Transmit at a very high frequency for better resolution.

Search radars scan a volume to *detect* targets. Once a target is detected and designated, a tracking radar's sole purpose is to follow that specific target and provide continuous, high-precision data on its position (range, azimuth, elevation).

4. The "servo system" in a mechanical tracking radar is responsible for:

◯ Generating the high-power transmit pulses.

◯ Filtering noise from the received signal.

◯ Converting electrical error signals into physical antenna movement.

◯ Calculating target velocity from Doppler shift.

The servo system (servo amplifiers and motors) acts as the controller and actuator. It takes the low-power error voltage from the discriminator, amplifies it, and drives the antenna's azimuth and elevation drive motors to move the antenna in the direction that minimizes the error.

5. According to Kepler's laws, a satellite in a circular orbit has an orbital period primarily determined by ....

The satellite's mass

The satellite's transmit power

The semi-major axis of its orbit

The inclination of its orbit

The semi-major axis of its orbit: Kepler's Third Law states that the square of the orbital period is proportional to the cube of the semi-major axis of the ellipse. For a circular orbit, the semi-major axis is simply the orbital radius. Mass is not a factor (for a satellite mass negligible compared to Earth).

5. The key parameter that sets the *theoretical maximum* data rate for a digital satellite channel in the presence of noise is:

a) EIRP (Effective Isotropic Radiated Power).

b) Bandwidth and Signal-to-Noise Ratio .

c) Transponder saturation flux density.

d) Antenna beamwidth.

Bandwidth and Signal-to-Noise Ratio (Shannon's Theorem): Shannon-Hartley Theorem states the channel capacity $C = B * \log_2(1 + SNR)$. This is the fundamental limit, regardless of other system parameters like EIRP, which affect the achievable SNR.

6. An electrical engineer is sizing the power subsystem for a LEO satellite. Which factor presents the most significant challenge compared to a geostationary (GEO) satellite?

◯ Higher power transmission requirements due to the longer average distance to ground stations.

◯ Frequent and relatively long eclipse periods when the satellite is in Earth's shadow, requiring robust battery systems.

◯ The need for nuclear power sources due to insufficient solar flux.

◯ Constant, uninterrupted solar illumination, leading to thermal over-control issues

A key design challenge for LEO satellites is their frequent passage through Earth's shadow (eclipse). These eclipses can occur for about 30-40 minutes per ~90-minute orbit. During this time, the satellite must run on batteries, which must be sized to handle this regular, deep discharge cycle, unlike GEO satellites which experience only brief eclipse seasons.

. What is the approximate orbital period of a LEO satellite?

◯ 90-120 minutes

◯ 24 hours

◯ 12 hours

◯ d) 60 minutes

Due to their low altitude, LEO satellites complete an orbit around Earth in about 90 to 120 minutes, traveling at speeds of about 7.8 km/s.